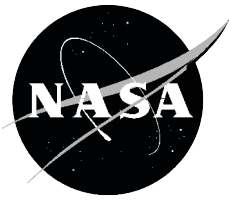


NASA/TM-2010-214655



# **2008 and 2009 Research and Engineering Annual Report**

*Bradley C. Flick and David F. Voracek  
NASA Dryden Flight Research Center  
Edwards, California*

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**May 2010**

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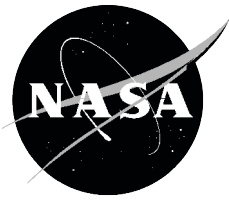
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NASA Dryden Flight Research Center  
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*National Aeronautics and  
Space Administration*

*Dryden Flight Research Center  
Edwards, California 93523-0273*

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### **Research Engineering Directorate**

Director – Bradley C. Flick  
Deputy Director – David F. Voracek  
Administrative Officer – Yvonne D. Kellogg

# AEROELASTIC UNCERTAINTY ANALYSIS TOOLBOX

## Summary

Flutter is a potentially explosive phenomenon that results from the simultaneous interaction of aerodynamic, structural, and inertial forces. The nature of flutter requires that flight-testing be cautious and conservative. In addition to flutter instability, adverse aeroelastic phenomena include limit cycle oscillations, buffeting, buzz, and undesirable gust response. The analytical prediction of aeroelastic phenomena in the transonic regime has historically been troublesome and requires high fidelity simulation models to obtain accurate predictions. These models are, however, computationally expensive. Traditional uncertainty analysis is therefore not often applied to flutter prediction. In the present research, computationally efficient methods that reduce the existing computational time limitations of traditional uncertainty analysis have been demonstrated. Specifically, this work has successfully demonstrated the coupling of Design of Experiments (DOE) and Response Surface Methods (RSM) and the application of robust stability techniques, namely  $\mu$ -analysis. These innovations have been shown to be more efficient while retaining accuracy. The innovations also have the flexibility to use computational unsteady aerodynamic and structural finite element models from a variety of sources, ranging from simple potential flow models (for example, doublet lattice methods) and linear structural models to solutions based on modeling of the full Navier-Stokes equations and non-linear structural models with many elements.

## Objective

The overall objective of this research is to demonstrate the feasibility of building an Aeroelastic Robustness Toolbox and to interface it with a state-of-the-art nonlinear aeroelastic code (AERO, developed by CMSOFT, Inc., Palo Alto, California). Initially, a “baseline” uncertainty analysis was performed using traditional stochastic simulation (Monte Carlo simulation) and an aeroelastic model of the AGARD 445.6 wing. Results from this analysis were used as a basis for evaluating the proposed uncertainty approaches. The application of DOE and RSM was applied to significantly reduce the number of simulation runs. A robust stability approach ( $\mu$ -analysis) was also applied to the aeroelastic problem to determine robust flutter boundaries. Additionally, an innovative approach was developed that includes aspects of, DOE and RSM as well as  $\mu$ -analysis. Finally, aspects of another uncertainty analysis technique, polynomial chaos, were explored.

## Approach

The approach to developing a prototype software toolbox that can be used for multi-method and rapid aeroelastic and aeroservoelastic uncertainty analysis of aerodynamic and structural models that span all degrees of complexity is as follows: develop, evaluate, and refine the innovative methods featured in the toolbox that address uncertainty robustness, including DOE and RSM as well as  $\mu$ -analysis techniques (fig. 1); develop and validate the computational tools using a state-of-the-art, high-fidelity, combined aerodynamic and structural code; incorporate feedback control into the toolbox as a means to conduct robust aeroservoelastic performance analysis; explore the integration of other capabilities with the toolbox including the National Aeronautics and Space Administration (NASA) flutterometer tool developed at the Dryden Flight Research Center (Edwards, California).

## Status

The three uncertainty methods (Monte Carlo, DOE and RSM, and  $\mu$ -analysis) have been effectively demonstrated for use with an aeroelastic wing model. The validity of the DOE and RSM method has been effectively demonstrated, as the results are almost identical to the baseline full Monte Carlo analysis. The computational time advantage of the DOE and RSM method has also been successfully illustrated by the fact that the number of runs of the full model has been reduced by two orders of magnitude and the computational time has been reduced 50 fold. The  $\mu$ -analysis method has been shown to produce valid results. Robust flutter boundaries found using the  $\mu$ -analysis approach have been shown to compare with the Monte Carlo results very well. Robust flutter boundaries found using the  $\mu$ -analysis approach are

expectedly and slightly more conservative, and should, in effect, be better since  $\mu$ -analysis represents an infinite-run Monte Carlo case. The DOE and RSM method has also been used to complement the  $\mu$ -analysis method very well and the work conducted has effectively demonstrated the union of these two techniques. The feasibility of including these multiple uncertainty methods into a comprehensive toolbox has been effectively demonstrated for use with aircraft models of varying complexity.

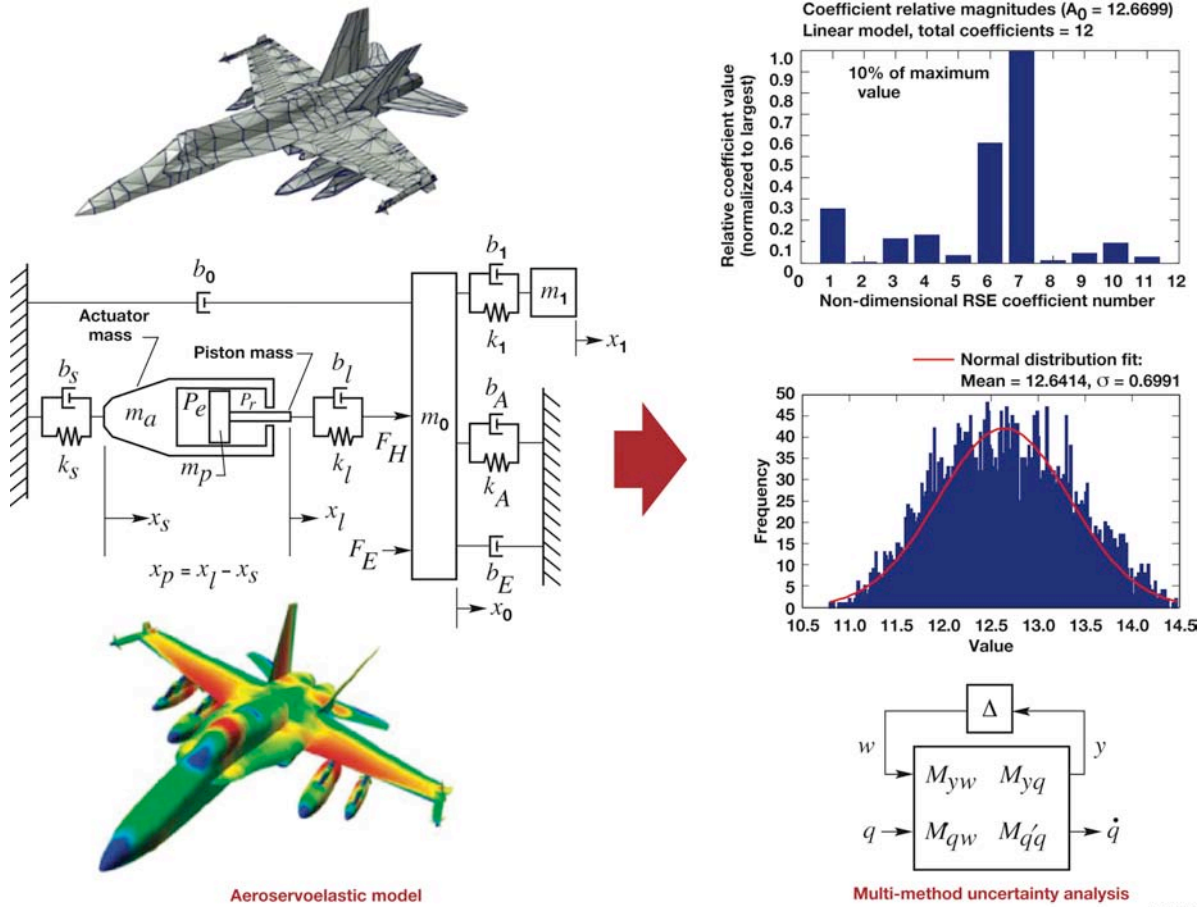


Figure 1. Aeroservoelastic modeling relation to multi-method uncertainty analysis.

### Applications

Enhanced predictive flutter boundaries will decrease the potential to test beyond the flutter boundary. This will decrease the risk of unexpected flutter and thereby increase flight safety. Rapid uncertainty analysis capability will be highly beneficial during aeroservoelastic flight tests to more accurately predict the flight envelope prior to testing. Applying the uncertainty analysis early in the design process will enable manufacturers to design and certify aircraft with expanded flight envelopes that are robust to uncertainties pertaining to aeroelastic phenomena such as flutter. Furthermore, rapid uncertainty analysis capability will decrease the amount of incremental flight-testing, thereby reducing development costs.

### Contacts

Martin J. Brenner, DFRC, Code RS, (661) 276-3793  
Systems Technology, Inc., Hawthorne, California, (310) 679-2281



# **DISTRIBUTED AERODYNAMIC SENSING AND PROCESSING TOOLBOX: APPLICATION WITH THE AEROSTRUCTURES TEST WING**

## **Summary**

Recent studies demonstrate a direct correlation between the unsteady aerodynamic forces and the instantaneous spatial locations of a few critical aerodynamic flow feature indicators (CAFFI) such as the leading-edge stagnation point (LESP) and flow separation point (FSP) using flush-mounted, micron-thin hot-film sensor arrays. These experiments have demonstrated that CAFFI can be used as an unsteady aerodynamic “observable” in the same manner that strain gages and accelerometers are used to measure structural response. Furthermore, since the changes in the instantaneous spatial location of CAFFI are directly related to the changes in the aerodynamic forces and moments, these “observables” can be used to initiate control actuation to realize desired load distribution and flight mission. Wind-tunnel experiments have demonstrated that:

1. The LESP can be used to determine the variations in the instantaneous (unsteady) lift generated by a wing section in the presence of gusts as well as structural oscillations.
2. The instantaneous lift curve slope could be obtained using the LESP even without a priori calibration of the sensors.
3. Absolute values of the lift coefficient in unsteady flow are obtained as a function of the instantaneous locations of the LESP and the FSP.

## **Objective**

The proposed experiment will extend these advanced concepts to flight research and development (R&D) applications. This effort will be devoted to minimizing the number of sensors for accurate real-time determination of CAFFI and optimization of hardware size, weight, and power requirements for applications using the aerostructures test wing 2 (ATW2) mounted on the F-15B (McDonnell Douglas Corporation, now The Boeing Company, Chicago, Illinois) Aero Flight Test Fixture (AFTF).

This flight test will be the first of its kind to measure unsteady aerodynamic loads (forcing function) in real time and correlate them with the structural response. The ATW2 test article will be used to characterize structural dynamic and aerodynamic behavior across a range of flight conditions, from low angle of attack to high angle of attack, from low Mach numbers to high Mach numbers, and in steady and unsteady maneuvers. Strain gages and accelerometers will be used to measure the structural response while hot-film gages will be used to characterize the aerodynamic flow features and to determine the aerodynamic forcing function. The flight experiment is expected to pave the way for the development of advanced computational modeling, flutter prediction techniques, and adaptive closed-loop control technology required for the design and development of flight vehicles with active aeroelastic wings.

## **Approach**

The plan is to design, build, and validate a distributed aerodynamic sensing and processing (DASP) toolbox using aerodynamic “observables” for flight R&D applications, and involves:

1. Designing and fabricating a flight-hardened aerodynamic sensing technology with deterministic real-time capabilities,
2. Quantifying the unsteady aerodynamic environment using the minimum number of sensors distributed along the wing span and chord,
3. Conducting design studies for power consumption, size, and weight requirements.

## Status

This work to date basically involved the design and fabrication of a prototype DASP toolbox including sensors with signal conditioners and signal processing software and hardware, flight system integration, and flight-test support with the following tasks:

1. Design and fabrication of the DASP toolbox measurement electronics,
2. Development and implementation of the digital aerodynamic signal processor,
3. Development and validation of compensation techniques,
4. Flight ruggedization and system integration,
5. Flight-test support and readiness,
6. Conduct of flight validation tests from October through December of 2009.

Figure 1 shows the DASP hot-films at the wing leading edge and at the leading tip of the wingtip probe (covered), with Macro-Fiber Composite (developed at the NASA Langley Research Center, Hampton, Virginia) actuators to be used for structural excitation. Accelerometers and strain gages have been installed, and a loads test and ground vibration test have been completed. Design optimization and integration, and ruggedization of sensors, signal conditioners, and signal processing components for the DASP toolbox is complete. Macro-Fiber Composite actuator control development and aircraft integration is in progress.



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Figure 1. Sensor distribution on the ATW2.

## Contacts

Martin J. Brenner, DFRC, Code RS, (661) 276-3793  
Christine V. Jutte, DFRC, Code RS, (661) 276-3058  
Tao of Systems Integration, Inc., Hampton, Virginia (757) 220-5040

# **REDUCED UNCERTAINTIES IN THE ROBUST FLUTTER ANALYSIS OF THE AEROSTRUCTURES TEST WING**

## **Summary**

Improved and/or accelerated decision making is a crucial step during flutter certification processes. Unfortunately, most of finite element models have uncertainties associated with model validity. Tuning the finite element (FE) model using measured data to minimize the model uncertainties is a challenging task in the area of structural dynamics. The test validated finite element model can provide more reliable flutter speed to define the flutter placard speed to which the aircraft can be flown prior to flight flutter testing. Having accurate rigid body dynamics is important for flight control system design and aeroelastic trim analysis. In the model tuning process, one requires not only satisfactory correlations between numerical and experimental results, but also the retention of mass and stiffness properties of the structures.

## **Objective**

The primary objective of this study is to reduce uncertainties in the structural dynamic FE model of the aerostructures test wing 2 (ATW2) to increase the safety of flight. Discrepancies are common between the test data and numerical results. However, the FE model can be fine tuned through the use of ground vibration test (GVT) data. Accurate and reliable GVT results are important to this adjusting process.

The secondary objective of the current study is to add model-tuning capabilities in the National Aeronautics and Space Administration (NASA) Dryden Flight Research Center (DFRC) (Edwards, California) object oriented Multidisciplinary Design, Analysis and Optimization (MDAO) Tool. The object oriented MDAO tool development was supported mainly by Aeronautics Research Mission Directorate (ARMD) Subsonic Fixed Wing project and partly by ARMD Supersonics project under the Fundamental Aeronautics program. This model tuning technique is essentially based on a non-linear optimization problem.

## **Approach**

A block diagram for a robust flutter analysis procedure used at NASA DFRC is given in figure 1. Using the FE model for a structural dynamic analysis becomes increasingly important in modern aircraft design and analysis processes. Generally, the quality of the initial FE model of an aircraft is not guaranteed, thus, GVTs are performed to validate the model. Most newly built FE models require tuning to minimize uncertainties in the structural dynamics FE models and flutter boundary results. Robust flutter analyses performed at NASA DFRC are mainly based on these validated FE models as shown in figure 1.

Model tuning is a common method to improve the correlation between numerical and experimental modal data, and many techniques have been proposed. These techniques can be divided into two categories: direct methods (adjust the mass and stiffness matrices directly) and parametric methods (correct the models by changing the structural parameters). Direct methods correct mass and stiffness matrices without taking into account the physical characteristics of the structures and may not be appropriate for use in model tuning processes. This summary discusses the parametric tuning method. In the optimization process, structural parameters are selected as design variables: structural sizing information (thickness, cross sectional area, area moment of inertia, torsional constant, etc cetera); point properties (lumped mass, spring constants, et cetera); and materials properties (density, Young's modulus, et cetera). Objective function and constraint equations include mass properties, mass matrix orthogonality, frequencies, and mode shapes. The use of these equations minimizes the difference between numerical results and target data.

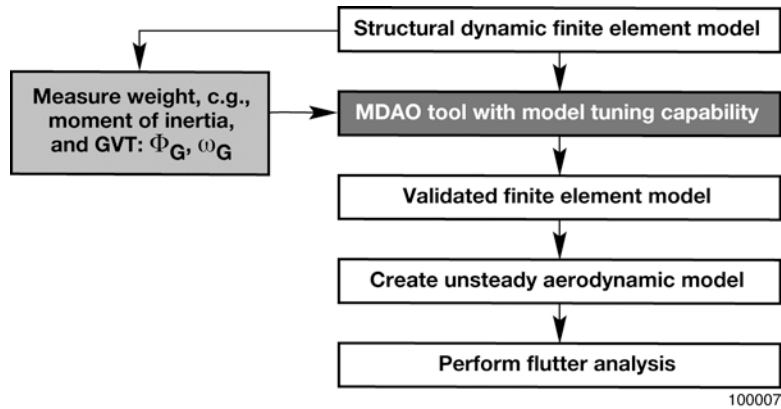


Figure 1. Robust flutter analysis procedure at NASA Dryden Flight Research Center.

### Status

Flutter boundaries of the ATW2 before and after model tuning are compared with the flight envelopes as given in figure 2. The green solid lines and the green dashed lines show the flight envelope of the mother ship and its 15 percent margin, respectively. The black solid line represents the planned ATW2 test envelope, and the gray dashed line is the 15 percent margin of the ATW2 test envelop. The red line with the square markers shows the flutter boundary before model tuning, and the blue line with the square markers represents the flutter boundary after model tuning. It should be noted that flutter boundary after model tuning is more conservative than before model tuning.

Without reduced uncertainties, 40 percent of the flutter margin (15 percent per military specifications plus 25 percent of flutter speed error) is needed. Therefore, with the updated FE model, the accuracy of flutter analysis can be improved and the flutter boundary prediction will be more reliable.

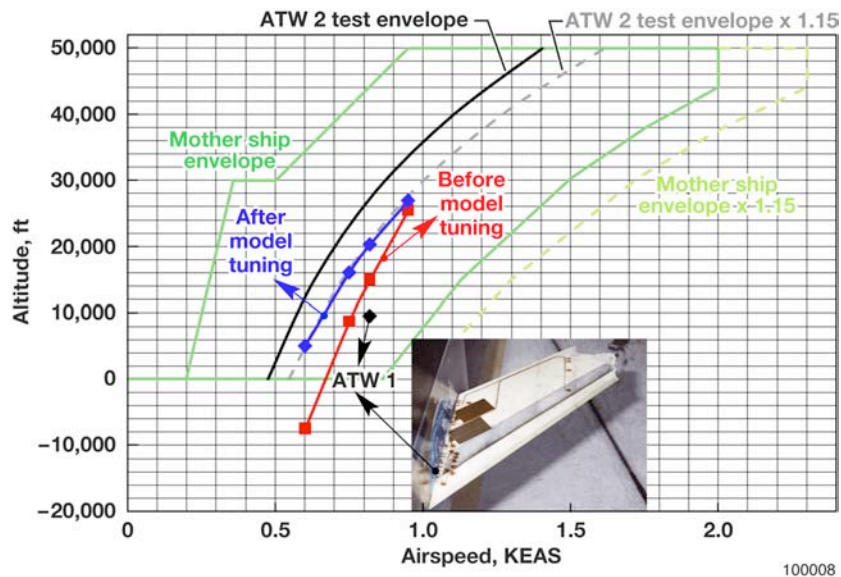


Figure 2. Flutter boundary before and after model tuning.

### Contacts

Dr. Chan-gi Pak, DFRC, Code RS, (661) 276-5698  
 Dr. Shun-fat Lung, DFRC, TYBRIN Corporation, Code RS, (661) 276-2969

# AEROSERVOELASTIC STABILITY ANALYSIS OF THE AEROSTRUCTURES TEST WING USING AUTOMATIC LOOP-BREAKING TEST

## Summary

Historically, analytical and experimental investigations concerning the interaction of the elastic airframe with aerodynamic and inertia loads have been conducted during the design phase to determine the existence of aeroelastic instabilities, so called “flutter.” With the advent and increased usage of flight control systems, there is also a likelihood of instabilities caused by the interaction of the flight control system and the aeroelastic response of the aircraft, known as aeroservoelastic instabilities. A diagram of the aeroservoelastic system is shown in figure 1.

Flutter safety is mandatory for all aircraft. Due to the lack of proper tools for analyzing these flutter safety margins, determination of the aeroservoelastic stability is a time consuming procedure, and the root locus method has been used at the National Aeronautics and Space Administration (NASA) Dryden Flight Research Center (DFRC) (Edwards, California). Recently, an automatic loop-breaking test for the computation of flutter safety margins (figs. 2–6) was formulated and incorporated into the in-house aeroservoelastic stability analysis code.

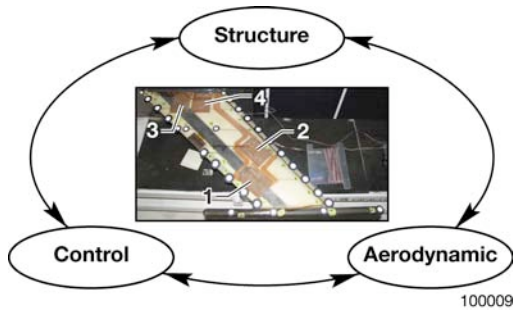


Figure 1. Aeroservoelastic system.

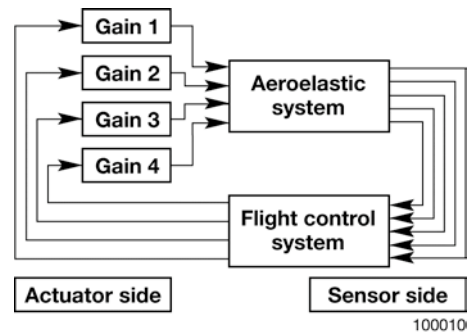


Figure 2. Schematic diagram of the aeroservoelastic system.

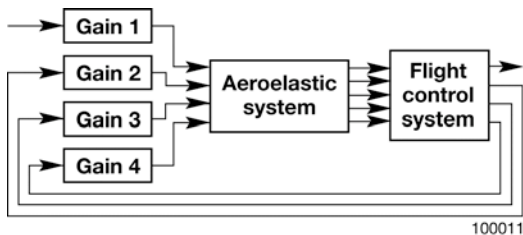


Figure 3. Loop breaking at actuator number 1.

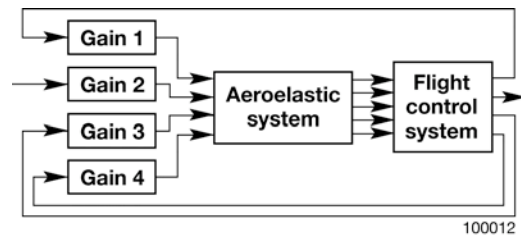


Figure 4. Loop breaking at actuator number 2.

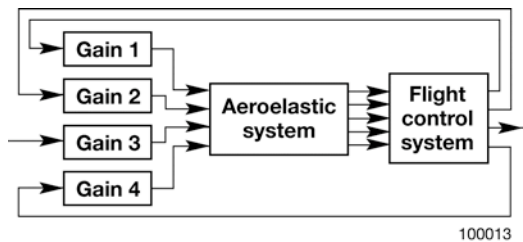


Figure 5. Loop breaking at actuator number 3.

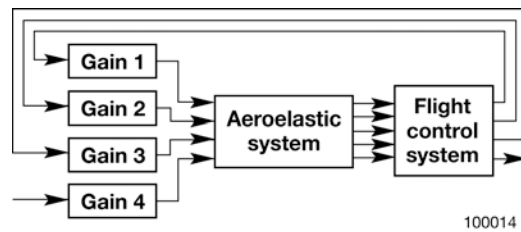


Figure 6. Loop breaking at actuator number 4.

## Objective

The primary objective of this study is to perform a linear flutter and aeroservoelastic analyses using the in-house aeroservoelastic stability analysis code based on the loop-breaking test.

## Approach

It should be noted that the gain and phase margins are defined only for a single-input single-output (SISO) system. The basic idea of the loop-breaking test is shown in figures 2–6. When the close loop multi-input multi-output (MIMO) aeroservoelastic system as shown in figure 2 is broken at actuator number 1, then this MIMO system becomes an open loop SISO aeroservoelastic system. The block diagram for this system is given in figure 3. When the loop-breaking is applied at actuators number 2, 3 and 4, then the block diagram given in figure 2 becomes figures 4, 5 and 6, respectively.

Once the SISO system is obtained, then the gain and phase margins can be calculated using the results obtained from the Bode diagram. The Bode diagram of an example problem is shown in figure 7. The gain and phase margins are easily evaluated from the Bode diagram. For the log magnitude-phase diagram, critical stability points are the 0 dB gain and -180 degree phase.

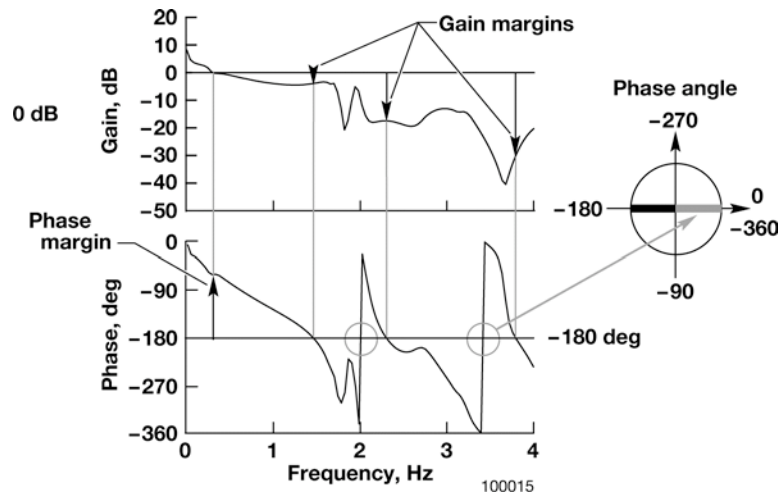


Figure 7. Computations of gain and phase margins using a Bode diagram.

A test article called the aerostructures test wing (ATW) with four piezoelectric patches, as shown in figure 1, has been built for demonstration of state-of-the-art sensor technologies for simultaneous, distributed, collocated measurement of shear stress (skin friction), steady and unsteady pressures, and structural strain and accelerations for mode shapes and other modal properties. This wing will also be used for active-adaptive flexible motion control studies, such as active vibration and flutter suppressions. In this active suppression study, the four piezoelectric patches will be used as the actuators.

## Status

The abstract submitted to the 27th Congress of the International Council of the Aeronautical Sciences at Nice, France 2010 was accepted. However, due to manpower and budget problems this paper was withdrawn.

## Contacts

Dr. Chan-gi Pak, DFRC, Code RS, (661) 276-5698  
Dr. Shun-fat Lung, DFRC, TYBRIN Corporation, Code RS, (661) 276-2969

# APPLICATION OF APPROXIMATE UNSTEADY AERODYNAMICS FOR DESIGN OPTIMIZATION

## Summary

A technique for approximating the modal Aerodynamic Influence Coefficient (AIC) matrix by using basis functions has been developed, and a process for using the resulting approximated modal AIC matrix in aeroelastic analysis and design optimization has been developed. The technique presented has been shown to offer accurate flutter speed prediction on the aerostructures test wing 2 (ATW2) and a Hybrid Wing Body (HWB) type of vehicle configuration with negligible loss in accuracy.

## Objective

There is considerable motivation to be able to perform aeroelastic calculations more quickly and inexpensively. The primary goal behind the current development of using an approximated modal AIC matrix based on the Basis Function Approach (BFA) for the rapid computation of a modal AIC matrix and aeroelastic response is to reduce the computation time for generating AIC matrices during the optimization procedure.

## Approach

The steps of the approximation process are depicted in the flowchart given in figure 1. In step 1, a set of representative basis functions,  $\Psi$ , are defined and intended to capture salient features of the modal responses the airplane is expected to encounter in the various design space. Each mode shape of the airplane with the design configuration can be represented as a linear combination of a set of the basis functions.

In step 2, the representative basis AIC matrices as shown in equation (1) are computed corresponding to the representative basis functions defined in step 1 at any Mach number and reduced frequencies .

$$\tilde{Q} = \Psi^T A \Psi \quad (1)$$

In step 3, for a set of given structural mode shapes  $\phi$ , each mode shape is decomposed in a linear combination of the basis functions  $\Psi$ . The  $i$ -th mode shape  $\phi_i$  ( $i=1, 2, \dots, m$ ) is approximated through the use of a least squares fitting together with the following basis functions as shown in equation (2),

$$\phi_i \approx \sum_{k=1}^n \beta_k^i \psi_k \quad (2)$$

where  $\psi_k$  is the  $k$ -th basis function and a coefficient  $\beta_k^i$  is the modal participation factors of the  $k$ -th basis function on the  $i$ -th mode shape,  $m$  is the number of mode shapes and  $n$  is the number of basis functions.

In step 4, an approximate modal AIC matrix  $\bar{Q}$  is computed based on a basis AIC matrix  $\tilde{Q}$  and modal participation factors as shown in equation (3).

$$\bar{Q}_{ij} = \sum_{r=1}^n \sum_{s=1}^n \beta_s^i \beta_r^j \tilde{Q}_{sr} \quad (3)$$



The basis AIC matrix  $\tilde{Q}$  in equation (1) can be computed and saved before starting optimization as mentioned in step 2. During optimization, mode shapes,  $\phi_i$  s ( $i=1, 2, \dots, m$ ), are fitted using basis functions  $\psi_k$  and modal participation factors  $\beta_k^i$  as shown in equation (2), and the approximate modal AIC matrix  $\bar{Q}$  can be computed using equation (3).

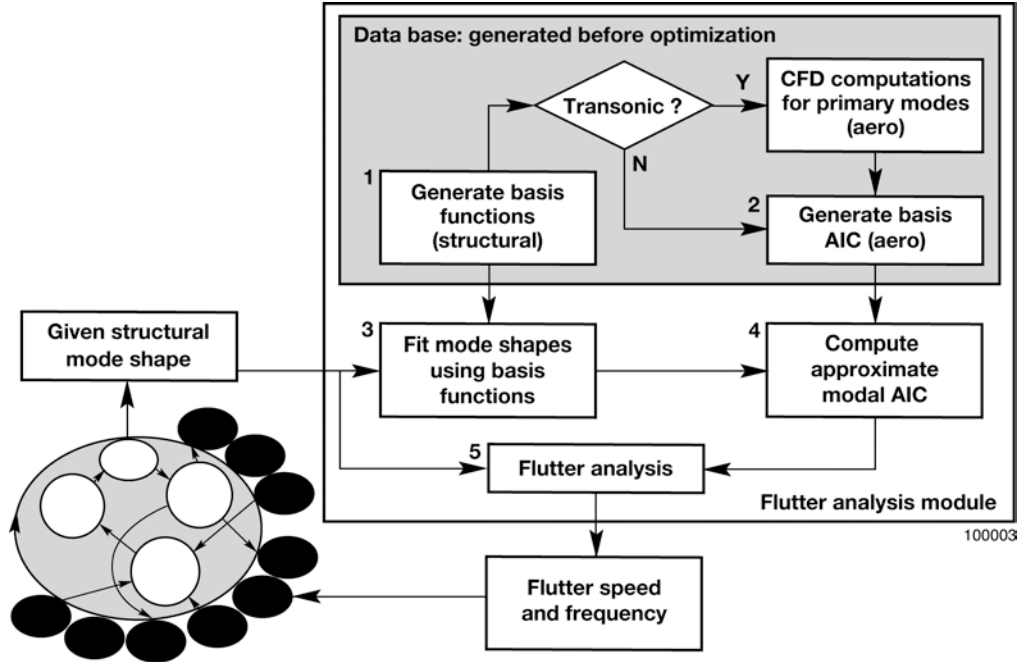


Figure 1. Flowchart of the flutter analysis module in the object-oriented MDAO tool.

### Status

A technique for approximating the modal AIC matrix by using basis functions has been proposed, and a process for using the resulting AIC matrix in aeroelastic analysis and design optimization has been developed and verified. The approximation method has been applied to the aeroelastic analyses, and the results are essentially identical to those using direct solution. The technique presented has been shown to offer accurate flutter speed prediction on a modified ATW2 configuration and a HWB type vehicle with negligible small loss in accuracy. These results may have practical significance in the analysis of aircraft aeroelastic calculation and could lead to a more efficient design optimization cycle.

The basis functions approach yields significant improvements in computational efficiency as compared to the original approach, thereby meeting the objective.

The abstract submitted to the 51st AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials conference was accepted and the final manuscript is being prepared.

### Contacts

Dr. Chan-gi Pak, DFRC, Code RS, (661) 276-5698  
Wesley W. Li, DFRC, Code RS, (661) 276-3138



# BASIS FUNCTION APPROXIMATION OF TRANSONIC AERODYNAMIC INFLUENCE COEFFICIENT MATRIX

## Summary and Objective

Designing today's aircraft for transonic speed is challenging because of the amount of computation time required for the unsteady aeroelastic analysis using computational fluid dynamics (CFD) code. Design approaches for this speed regime are largely based on manual trial and error; the time required for unsteady CFD computations considerably slows the entire design process. Analyses are usually performed repeatedly to optimize the final design. There is considerable motivation to be able to perform aeroelastic calculations more affordably for the transonic speed regime. The primary objective of the current study is to introduce a new approximation methodology to be used for the transonic aeroelastic optimization studies.

## Approach

The unsteady transonic aeroelastic analysis method for design optimization using unsteady aerodynamic approximation is being developed. The main conceptual innovation in this approach is to perform aeroelastic calculations more quickly and inexpensively. This approximation methodology will be integrated into the multidisciplinary design, analysis, and optimization (MDAO) tool for design optimization at a reasonable computational cost. A process, which efficiently incorporates approximated Aerodynamic Influence Coefficient (AIC) matrix into the structural optimization process has been developed and is outlined in the flowchart shown in figure 1.

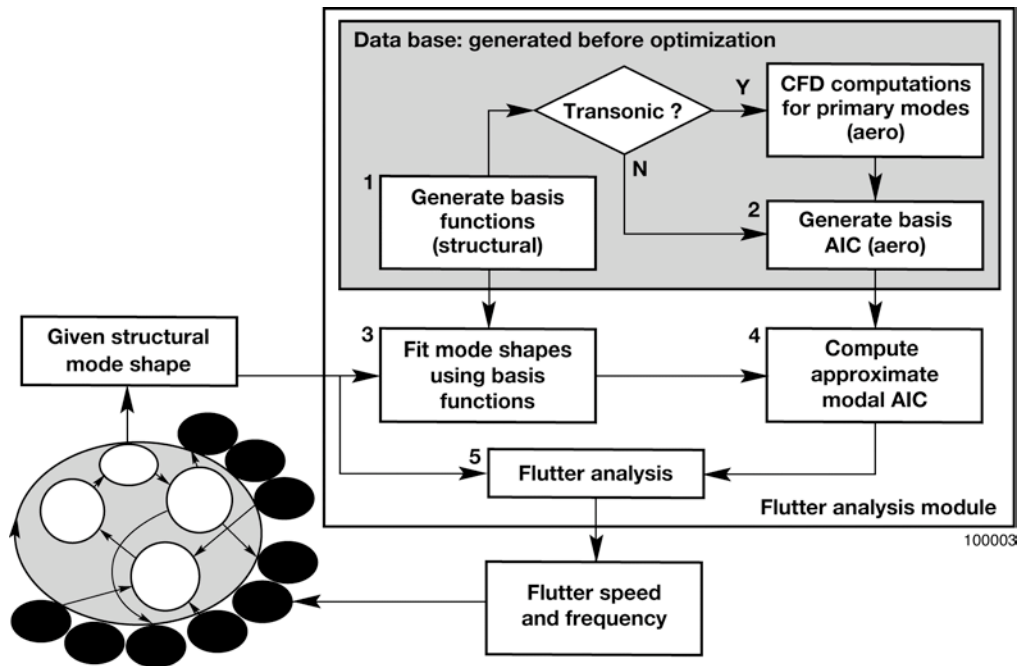


Figure 1. Approximate transonic flutter analysis flowchart.

This approximation method requires a set of representative basis functions based on structural mode shapes and the unsteady aerodynamic to be represented in the frequency or Laplace domain. Dynamically linear assumption is used for creating AIC matrices in transonic speed regime. Unsteady CFD computations are needed for the important columns of an AIC matrix which corresponded to the primary modes for the flutter. The transonic flutter can be found by the classic methods, such as rational function approximation, p-k, p, root-locus etc.

### Status

The proposed technique will be verified using the aerostructures test wing 2 designed, built, and tested at the NASA Dryden Flight Research Center (Edwards, California) and shown in figure 2. The resulting flutter solution using the approximate method will be compared with the direct flutter analysis.



Figure 2. The aerostructures test wing 2.

The abstract submitted to and accepted for the 27th Congress of the International Council of the Aeronautical Sciences at Nice, France 2010, and the final manuscript is being prepared.

### Contacts

Wesley W. Li, DFRC, Code RS, (661) 276-3138  
Dr. Chan-gi Pak, DFRC, Code RS, (661) 276-5698

# **UPDATING FINITE ELEMENT MODEL OF THE AEROSTRUCTURES TEST WING USING GROUND VIBRATION TEST DATA**

## **Summary**

A test article called the aerostructures test wing (ATW) was developed and flown at the National Aeronautics and Space Administration (NASA) Dryden Flight Research Center (DFRC) (Edwards, California) on the NF-15B (McDonnell Douglas, now The Boeing Company, Chicago, Illinois) test bed aircraft as shown in figure 1 for the purpose of demonstrating and validating flutter prediction methods during flight. The first aerostructures test wing (ATW1), flown in 2001, was originally developed to directly address requests for better flight flutter test techniques by providing a functional flight-test platform. While the first series of tests was extremely successful, the minimum amount of instrumentation (structural accelerometers and strain gages) was chosen to satisfy the scope of the program. These sensors were limited in the capability to answer questions of aeroelastic interactions, sources of nonlinearity, physical mechanisms of aeroelastic coupling, and feedback dynamics between the structure and aerodynamics.

A second aerostructures test wing (ATW2), as shown in figure 2, was built for demonstration of state-of-the-art sensor technologies for simultaneous distributed collocated measurement of shear stress (skin friction); steady and unsteady pressures; and structural strain and accelerations for mode shapes as well as other modal properties. Like the ATW1, the ATW2 was flown on the NASA NF-15B airplane. In order to successfully predict the onset flutter, the structural dynamics finite element (FE) model must be robust and accurate. The ground vibration test (GVT) is used as one of the validation methods for determining the robustness of the FE model.

## **Objective**

The primary objective of this study is to obtain the GVT-validated structural dynamics FE model. Selection of measurement locations can be critical to the success of an experimental modal survey. Different sensor and exciter placement algorithms for pre-test evaluations were investigated to ensure the quality of the modal test.

An inefficient approach to correlate the FE model with test data is by manual trial-and-error methods. A more efficient approach is to use a mode matching technique for the model refinement of both ground and flight-based models. A model tuning technique utilized the NASA DFRC multidisciplinary design, analysis and optimization (MDAO) tool is used to adjust the structural properties such that the analytical results and the measured data are matched.

## **Approach**

It is important to assure that an adequate number of proper sensor locations are identified for the collection of data during the GVT. There are several existing techniques that can be used for the determination of measurement locations. Effective Independence; Genetic Algorithm (GA); Kinetic Energy Sorting; Guyan Reduction; Iterative Guyan Reduction and Driving Point Residues are sensor placement algorithms developed in this study.

Discrepancies in frequencies and mode shapes are minimized using a series of optimization procedures. There are two optimization algorithms adopted in the NASA DFRC MDAO tool: the traditional gradient-based algorithm, and the GA. Gradient-based algorithms work well for continuous design variable problems, whereas GAs can handle continuous as well as discrete design variable problems easily. When there are multiple local minima, GAs are able to find the global optimum results, whereas gradient-based methods may converge to a locally minimum value. In this research work, the GA is used for the solution of the optimization problem.

### Status

This work was presented at the 50th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials conference in Palm Springs, California, May 4-7, 2009.



Figure 1. Aerostructures test wing 1 mounted on the NASA NF-15B for flight flutter testing.

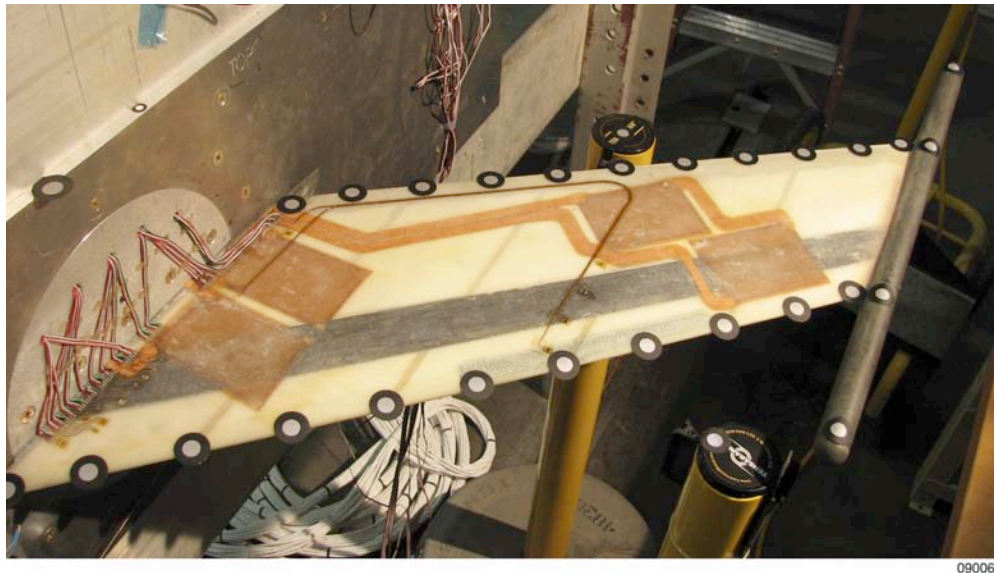


Figure 2. Aerostructures test wing 2.

### Contacts

Dr. Chan-gi Pak, DFRC, Code RS, (661) 276-5698  
Dr. Shun-fat Lung, DFRC, TYBRIN Corporation, Code RS, (661) 276-2969

# **CFD ANALYSIS OF LANCETS NOZZLE JET PLUME EFFECTS ON SONIC BOOM SIGNATURE**

## **Summary**

An axisymmetric full Navier-Stokes computational fluid dynamics (CFD) study is being conducted to examine nozzle exhaust jet plume effects on the sonic boom signature of a supersonic aircraft. A simplified axisymmetric nozzle geometry, representative of the nozzle on the National Aeronautics and Space Administration (NASA) Dryden Flight Research Center (DFRC) (Edwards, California) NF-15B (McDonnell Douglas, now The Boeing Company, Chicago, Illinois) Lift and Nozzle Change Effects on Tail Shock (LaNCETS) research airplane, is considered. The highly underexpanded nozzle flow is found to provide significantly more reduction in the tail shock strength in the sonic boom N-wave pressure signature than perfectly expanded and overexpanded nozzle flows. A tail shock train in the sonic boom signature, similar to what was observed in the LaNCETS flight data, is observed for the highly underexpanded nozzle flow. Axisymmetric CFD simulations show the flow physics inside the F-15 nozzle to be complex and multidimensional. The CFD results provide a detailed description of the nozzle flow physics involved in the LaNCETS nozzle at different nozzle expansion conditions and help interpreting LaNCETS flight data as well as in the eventual CFD analysis of a full LaNCETS aircraft.

## **Objective**

The primary objective of the current CFD research effort is to support the LaNCETS flight research data analysis effort by studying the detailed nozzle exhaust jet plume's imperfect expansion effects on the sonic boom signature of a supersonic aircraft.

## **Approach**

Figure 1 illustrates the primary flow physics present in the interaction between the exhaust jet plume shock and the sonic boom coming off of an axisymmetric body in supersonic flight. A structured finite-volume compressible full Navier-Stokes CFD code is used in the current study. This approach is not limited by the simplifying assumptions inherent in previous sonic boom analysis efforts using the linearized potential flow equation and the method of characteristics. Also, this study is the first known jet plume sonic boom CFD study in which the full viscous nozzle flow field is modeled, without coupling to a sonic boom propagation analysis code, from the stagnation chamber of the nozzle to the far field external flow, taking into account all nonisentropic effects in the shocks, boundary layers, and free shear layers, and their interactions at distances up to 30D from the jet centerline. To examine the jet plume's imperfect expansion effects on the sonic boom signature, a simplified axisymmetric nozzle geometry, representative of the nozzle on the NASA DFRC NF-15B LaNCETS research aircraft, is studied.

## **Status**

Axisymmetric CFD analysis was conducted to study the nozzle exhaust jet plume's imperfect expansion effects on the sonic boom signature of a supersonic aircraft. A CFD solution is shown in figure 2; the flow field is very complicated and multidimensional with shock-shock and shock-plume interactions. The highly underexpanded nozzle flow provided more reduction in the tail shock strength in the sonic boom pressure signature than the overexpanded, perfectly expanded, and mildly overexpanded nozzle flows. The far field shock train was present only in the highly underexpanded jet plume solution (fig. 2) and absent for all other cases. The current study also provided important information on modeling the LaNCETS aircraft nozzle in a future full-aircraft CFD study. Ongoing 3D CFD study is being conducted to evaluate the effects of nozzle vectoring. Also, we are collaborating with NASA Glenn to set up a supersonic wind tunnel test for the LaNCETS aircraft nozzle geometry. The results of the 3D CFD study will be compared and validated by the NASA Glenn Research Center (Cleveland, Ohio) supersonic wind tunnel test data as well as the LaNCETS flight data.

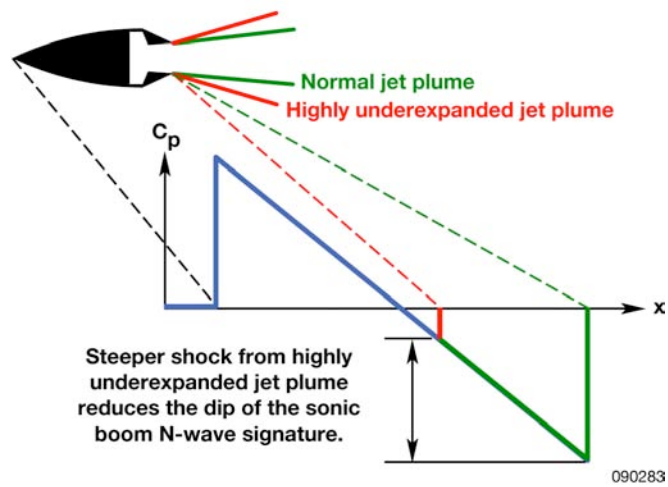


Figure 1. Jet plume shock–sonic boom interaction flow physics.

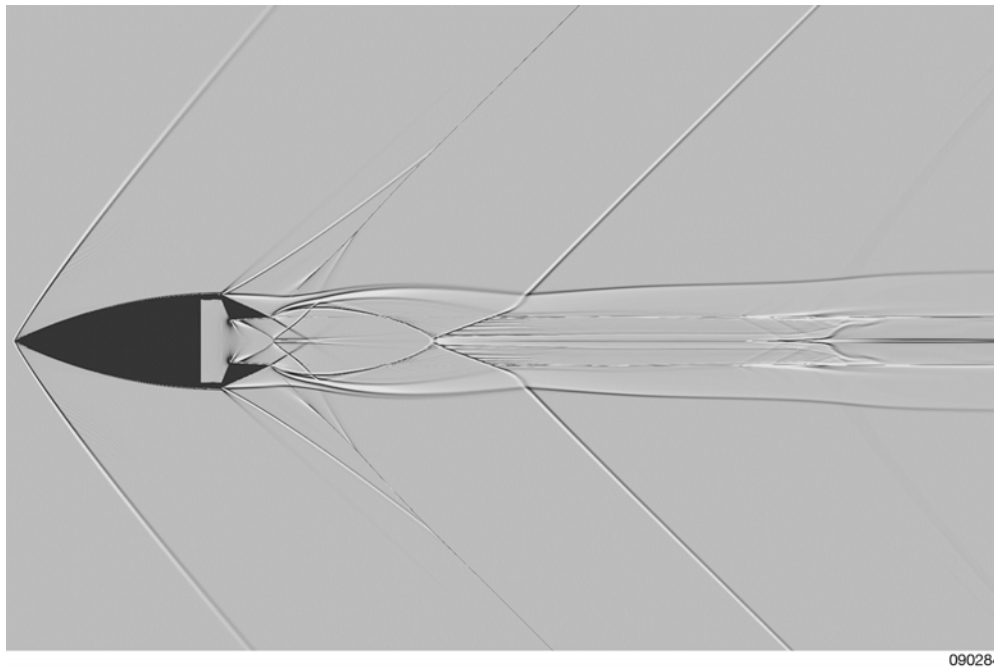


Figure 2. A CFD shadowgraph visualization of imperfectly expanded jet plume effects.

### Contacts

Trong Bui, DFRC, Code RA, (661) 276-2645



# USING INNER LOOP THRUST VECTORING CONTROL LAWS FOR LIFT AND NOZZLE CHANGE EFFECTS FLIGHT RESEARCH

## Summary

The NASA Dryden Flight Research Center (Edwards, California) successfully completed the Lift and Nozzle Change Effects on Tail Shock (LaNCETS) flight tests in January 2009 to investigate the tail shock region of an F-15 (McDonnell Douglas, now The Boeing Company, Chicago, Illinois) airplane by probing the shock waves around it, using another F-15 airplane. The two airplanes are shown in figure 1. Changes in lift distribution and plume shape were used to investigate the tail shock region. In order to change the lift distribution and plume shape of the F-15 airplane, a decade-old Inner Loop Thrust Vectoring (ILTV) research control law was used. Flight envelope expansion was performed for the test configuration and flight conditions prior to the probing test points. The flight conditions flown in support of LaNCETS are shown in figure 2.



EC05-0148-04

Figure 1. The NASA NF-15B-837 and F-15B-836 airplanes.

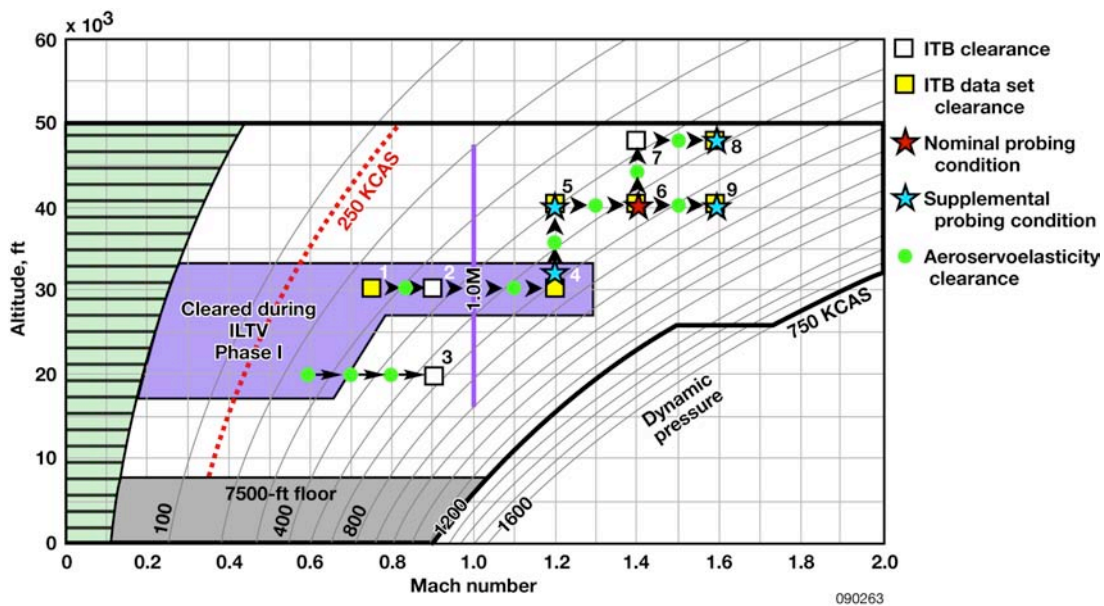


Figure 2. LaNCETS flight conditions.

## **Objective**

The purpose of the LaNCETS test is to investigate the effects of lift distribution and nozzle area ratio changes on tail shock strength. Specific research objectives are to: obtain in-flight shock strength for multiple combinations of nozzle area ratio and lift distribution, compare results with pre-flight prediction tools, and update predictive tools with flight results. The objectives from a stability and control perspective are to ensure adequate aircraft stability for the changes in lift distribution and plume shape, and ensure manageable transient from engaging and disengaging the ILTV research control laws.

## **Approach**

The approach for achieving the research objectives was to utilize the unique capabilities of the NASA NF-15B airplane, tail number 837, to allow the adjustment of the nozzle area ratio and canard positions by engaging the ILTV research control laws. The NASA F-15B airplane, tail number 836, equipped with a special sonic-boom-measuring noseboom, conducts near-field probing of the full-airplane shock structure to determine the nozzle and lift change effects on shock strength. Changes to the nozzle area ratio affect plume shape, while canard positions affect the lift distribution over the airplane. The ILTV control laws provide the ability to add trim command biases to canard positions, nozzle area ratios, and thrust vectoring through the use of “datasets.” Datasets consist of programmed test inputs (PTI) that define “trims” to change the nozzle area ratio and canard positions. The trims are applied as increments to the normally commanded positions.

A LaNCETS non-linear six-degrees-of-freedom simulation capable of real time pilot-in-the-loop, hardware-in-the-loop, and non-real-time batch support was developed and validated. Prior to the first flight, extensive simulation analyses were performed to show adequate stability margins with the changes in lift distribution and plume shape. Additionally, analysis of transient engagement and disengagement was also performed to show manageable transients.

A buildup approach was used for flight-testing to ensure safety. Flight envelope clearance and dataset checkout were performed in the subsonic flight regime before proceeding to the supersonic flight regime. Flight envelope clearance was tailored from the ILTV flight program, which consisted of raps and doublets maneuvers. These maneuvers were considered sufficient to ring out any aeroservoelasticity, stability, or structure loads with the ILTV research flight controller since probing was mainly level flight.

## **Contacts**

Cheng Moua, DFRC, Code RC, (661) 276-5327



# **ORION ABORT FLIGHT TEST CREW MODULE AND ADAPTER CONE SHAKER TEST**

## **Summary**

New acoustic/structure-borne environments obtained from the ST-1 Abort Motor test fire (Lockheed Martin, Bethesda, Maryland; and Alliant Techsystems, Inc., Minneapolis, Minnesota) showed that the thrust vector and center of gravity (C.G.) had changed for the Orion Abort Flight Test PA-1. Moving the thrust vector away from the C.G. caused increased loads of the primary structure. Vibroacoustic levels from the ST-1 test were approximately two times higher than those predicted and delivered to the Loads and Dynamics Group. These environments exceeded previous analysis input environments used to produce component design specifications and the finite element model showed negative margins. In an effort to study the effects of ST-1 environments, acoustic and vibration testing was performed on the partial Pad Abort One (PA-1) vehicle assembly (the Crew Module and the Adapter Cone) to quantify load-path-specific transfer functions, and vehicle damping.

In addition to the other mitigating efforts, testing was necessary to remove model and analysis uncertainties. The test data provided the PA-1 Coupled Loads Analysis version 6 (CLA-6) loads assessment team data to supplement, give guidance on application, or replace model predictions.

The results from the shaker test showed 1-2% damping exhibited in mid range (50-250 Hz) and lower damping (0.5%) in the high range (> 250 Hz). The final conclusion was that the 1% damping used in the analysis overall agreed with test data.

## **Objective**

A modal test on the Crew Exploration Vehicle (CEV) PA-1 was used to measure the damping and transfer functions of the structure, and allowed for more accurate modeling and measurement of the actual damping values. This test was conducted from June 16-20, 2009, to determine if the structural loads were less than (CLA-5B), and to give an understanding of negative margin and its associated risk.

## **Approach**

Transfer functions were predicted using the finite element model and specifically with the vehicle on top of a Dryden-designed soft support system, which isolates the vehicle from the ground, allowing a direct comparison of the vehicle predictions to the measured predictions from the shaker test. The shaker test provided transfer functions between the 500-lb shaker excitation and the accelerometers for the Loads and Dynamics Working Group. These transfer functions gave frequency content from 5-1000 Hz at low force levels and damping estimates were made from 50-1000 Hz at high force levels.

## **Shaker Test Configuration**

The assembled Crew Module and Adapter Cone were mounted on top of a Dryden-designed soft support system with a ball and cup interface. Without the soft support, the transfer functions would have needed to be adjusted to take out the effect of the fixed boundary condition. This was an extra step the project could not afford having such a short time in the schedule to apply the test data to the model.

A 500-lb shaker was suspended with a crane and attached to the top structural rim of the Adapter Cone (fig 1). A total of 134 accelerometer responses were recorded for 125 test runs. The test runs consisted of 4 force excitation configurations, 2 locations 90 degrees apart and each location having 2 orientations, vertical and 45 degrees, and many different force levels. Different orientations and locations of the 500-lb shaker were used to get the proper energy into all the accelerometer locations and to verify the symmetry of the vehicle load paths.



ED09-0102-719

Figure 1. Shaker test performed on the adapter cone and crew module at the NASA Dryden Flight Research Center (Edwards, California).

Equipment specifications on-board the crew module limited the maximum force excitation level. The highest random excitation force level was 119 lb root mean squared (RMS) and +/- 450 lb peak-to-peak. The highest sine-sweep excitation was 176 lb peak-to-peak at low frequencies and 197 lb peak-to-peak at high frequencies. The original force level used to create the predicted transfer functions was 500 lb. With the reduced shaker force levels, the data tended to be of poor quality.

### **Status**

The shaker test was completed on June 20, 2009 and the test results were provided to the Orion PA-1 Load and Dynamics Group (Lockheed Martin, Bethesda, Maryland; and Quartus Engineering Incorporated, San Diego, California). The results from the shaker test recommended a damping schedule of 2% critical damping at the low frequency range (0-50 Hz), 1% critical damping at the mid frequency range (150-250 Hz) and lower damping (0.5%) in the high frequency range (500-1000 Hz). The final conclusion was the 1% damping used in analysis overall agreed with the test data.

### **Contacts**

Sarr Ginn, DFRC, Code RS, (661) 276-3434

# **TESTING THE ORION (CREW EXPLORATION VEHICLE) LAUNCH ABORT SYSTEM ASCENT ABORT FLIGHT TEST**

## **Summary**

The National Aeronautics and Space Administration (NASA) has announced plans to retire the Space Shuttle fleet in 2010. The Constellation program was created by NASA to develop the next generation of manned space vehicles and launch vehicles. NASA's vision for the next manned spaceflight vehicle, known as Orion, involves a return to the capsule configuration similar to Mercury, Gemini, and Apollo. In the event of a catastrophic failure during the launch, the successful use of the launch abort system will ensure the safety of the crew. The abort flight tests are a series of four planned unmanned flight tests to verify the functionality of the launch abort system. The first ignition of the launch abort system from a dynamic condition will occur during the transonic ascent abort flight test. All abort flight tests are scheduled to take place at White Sands Missile Range (WSMR) in New Mexico. The transonic ascent abort launch is scheduled for the end of 2012.

## **Objective**

In the event of a catastrophic failure on the launch pad or during the ascending trajectory through first stage burnout, the Launch Abort System (LAS) will initiate an abort that will safely separate the crew module (CM) from the failed launch vehicle. The objective of the Orion abort flight tests is to examine the performance of the LAS through a series of four unmanned flight tests. Two of the flight tests that the Flight Test Office team has proposed are launch pad abort tests that will be performed from a launch stand instead of from an external booster. The remaining two proposed flight tests are ascent abort tests that will use an Abort Test Booster (ATB) to take the CM and LAS to a predetermined flight condition. The flight condition for the first planned ascent abort test is located at the transonic condition of the operational launch vehicle trajectory. The objective of transonic ascent abort is to demonstrate that the LAS has sufficient capability to propel the CM to a safe distance from a launch vehicle while in a high drag and dynamic environment.

## **Approach**

The ascent abort flight test vehicle is composed of an ATB, a separation ring, a CM, and an LAS. The CM for the ascent aborts will be representative of the operational CM, and is being built by Lockheed Martin Space Systems Company (Denver, Colorado) at NASA's Michoud Assembly Facility in New Orleans, Louisiana. The ATBs for the ascent aborts are being built by Orbital Sciences Corporation (Chandler, Arizona) and will be a single stage booster. The ATBs will not be representative of the two-stage booster for the operational launch vehicle, but will carry the CM and LAS to the appropriate test conditions along the operational launch vehicle trajectory.

The Flight Test Flight Dynamics team designs, analyzes, and validates the flight dynamics and trajectories of the flight test vehicle for all abort flight tests performed by the Flight Test Office. The first flight test is a pad abort, which will initiate from a static condition and will be travelling a much shorter distance than the ascent aborts. Since the LAS for an ascent abort is being initiated at a dynamic condition there is an increased level variability and complexity than from a pad abort. Some of the primary concerns are providing adequate separation distance between the CM and the ATB while staying within the structural limits and ensuring that all vehicle components and debris stay within the WSMR boundaries. The primary method of analysis involves creating six degree of freedom (DOF) Monte Carlo simulation runs. The Monte Carlo simulation runs are dispersed in atmospheric conditions, mass properties, parachute timing, aerodynamic uncertainties, and ATB separation conditions. The CM, LAS, ATB, and any other objects of significant mass are tracked from the launch pad until ground landing in the simulated environment. The results of the simulations are analyzed and adjusted to ensure that they fall within the constraints applied to the ascent abort flight tests. Figure 1 shows an illustrated example of the

sequence of events overlaid over a nominal ascent abort trajectory. Figure 2 shows the results from the six DOF Monte Carlo simulation of the crew module trajectory for an ascent abort at the maximum dynamic pressure condition of the operational launch vehicle trajectory. The NASA Dryden Flight Research Center (DFRC) (Edwards, California) team's work involves the generation of the simulated six DOF trajectories, analysis of the various vehicle sensitivities, and validation of simulations and models held by the various organizations involved with Orion Abort Flight Test.

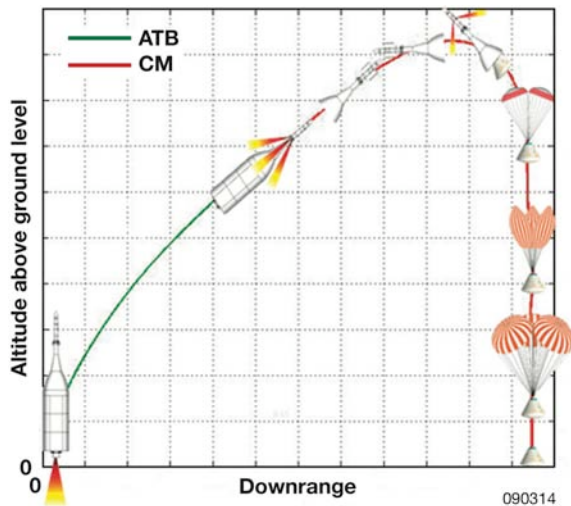


Figure 1. Sequence of events for an ascent abort flight test.

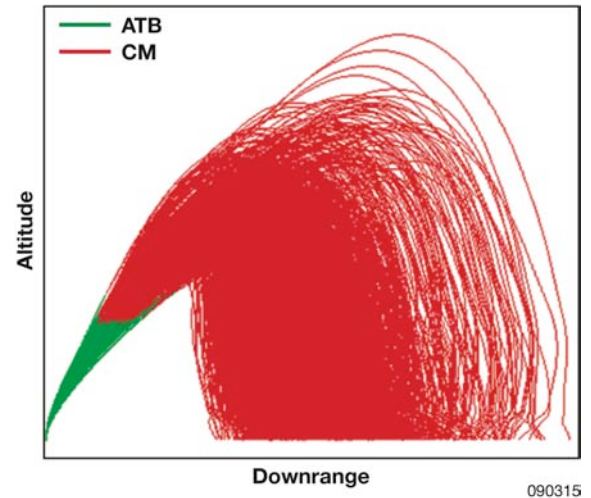


Figure 2. Crew module trajectories resulting from the six DOF Monte Carlo simulation runs of the maximum dynamic pressure ascent abort flight test.

### Status

All flight tests are scheduled to occur at White Sands Missile Range in New Mexico. The initial site selection for the transonic ascent abort is near launch complex 32E and is slated to be launched in a 37 degrees east of north direction relative to the launch pad at the end of 2012. The DFRC team has continued to refine the six DOF trajectories, developed simulation models, performed sensitivity analysis of the moments of inertia and aerodynamics, and performed studies on the ground impact area of the simulated bodies in various nominal and malfunction configurations.

### Contacts

Ryan Stillwater, DFRC, Code RC, (661) 276-3591  
Peggy Hayes, Flight Test Flight Dynamics lead, DFRC, Code RC, (661) 276-2508  
Bob Clarke, Flight Test Office lead engineer, DFRC, Code RC, (661) 276-3799

# **REAL-TIME DECOMPRESSION AND LOCAL MAP RENDERING FROM A HIGHLY COMPRESSED DIGITAL TERRAIN MAP FOR AN AUTOMATIC GROUND COLLISION AVOIDANCE SYSTEM**

## **Summary**

The automatic ground collision avoidance system (Auto-GCAS), the automatic air collision avoidance system (Auto-ACAS), and the automatic integrated air-and-ground collision avoidance system (Auto-ICAS) rely on sensors to detect a collision threat, algorithms to determine the potential and imminence of a collision, and an automatic pilot system (autopilot) to evade a potential collision. The Auto-GCAS uses a digital terrain map (DTM) as a primary input to determine ground proximity. Generally, the more accurately a DTM represents the actual terrain or the larger the area represented, the larger the DTM file size. Current Auto-GCAS designed for fighter-type aircraft utilize DTMs that are loaded on a flight-by-flight basis. This requires an interface for loading the data and introduces an operational risk of data being loaded incorrectly. Facilitating optimally infrequent DTM loading requires that the “gaming area” cover a much larger area than is typically used on aircraft such as the F-16 (Lockheed Martin, Bethesda, Maryland) Auto-GCAS (ideally a global area). For current DTM data formats the memory space required for such large-area DTMs is prohibitively large. Methods and software to significantly compress and also rapidly decompress DTM data were thus developed at the NASA Dryden Flight Research Center (DFRC) (Edwards, California) to facilitate the installation of large-region DTMs on aircraft with limited available memory and computational resources.

## **Objective**

The subject of this study was specifically the decompression and local map construction logic developed at DFRC. This software was developed to locally decompress the DFRC-compressed digital terrain maps (CDTMs) and to render a local terrain map in real time for an Auto-GCAS in a high-performance-type airplane.

## **Approach**

The current Auto-GCAS designed by Lockheed Martin (LM) utilizes a digital terrain map that is compressed to a limited extent and is decompressed to a grid format of elevation “posts” with a 6-arc-second resolution. The latitude and longitude of each elevation post is implied from the grid format. The Auto-GCAS scans terrain data in this format to determine a terrain profile, which is compared with a computed escape trajectory.

To enable implementation and flight-testing of the DFRC CDTM in the current F-16 Auto-GCAS, the decompression and local map generation method was developed to interface with the current LM terrain scanning implementation without requiring any significant modification to the LM Auto-GCAS software. This was done in a C++ environment. Because the DFRC CDTM is stored in a format very different from that of the current digital maps that are loaded into the Auto-GCAS, the decompression software decompresses the CDTM in a region local to the airplane position and then “rasterizes” a subset of the local decompressed data to match the same LM digital terrain grid format, but with a slightly higher resolution.

In order to satisfy the RAM availability limit requirement for a digital map in the Auto-GCAS advanced data transfer equipment (ADTE), only a very small part of the entire compressed map could be called into the program memory at any given time. The decompression and mapping software itself also had to fit within the program memory along with the output local raster data for scanning. For this reason, a local moving map containing both compressed and rasterized data was stored and updated in the program memory. The decompression process also had to be achieved in a way that minimized computational overhead to within the ADTE capability.

Beginning at initialization, the local map contains CDTM format data representing a 3-by-3 deg section of nine 1-by-1 deg cells of a geographic map and raster data outputted by the decompression logic representing a smaller 1.5-by-1.5 deg subsection of nine ½-by-½ deg cells within the 3-by-3 deg CDTM data section (fig. 1). Both sections of data are more or less centered on the aircraft position to allow rapid “rasterization” and scanning of data by the Auto-GCAS algorithm. These data sections are updated according to aircraft position whenever the aircraft position crosses defined boundaries. These boundaries (also called hysteresis boxes) are initialized and updated to be centered on the compressed data cell and the smaller raster data cell the aircraft position is in (fig. 1).

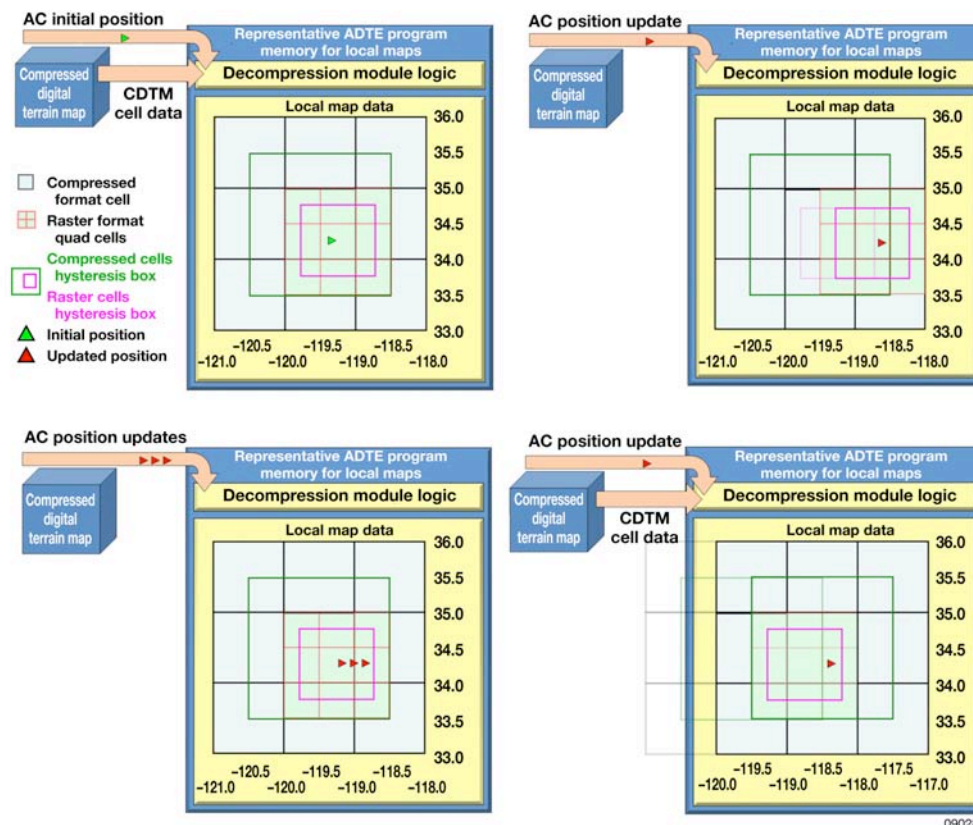


Figure 1. Local map construction process; the updating sequence occurs clockwise from top left.

### Status

The decompression software was designed, developed, documented, and successfully batch-tested at DFRC in a D-Six software simulation to verify correct decompression and local map updating, and to stress the software for high speeds over complex terrain. The software was delivered to LM on schedule for class B level integration and testing in an integrated software and advanced data transfer unit (ADTU) environment. Initial testing seems to indicate that the software executes and updates correctly and efficiently. Further testing and integration was postponed due to schedule constraints imposed from the primary Auto-GCAS flight test objectives.

### Contacts

Mark Skoog, DFRC, Code Z, (661) 276-5774  
 Loyd Hook, DFRC, Code RF, (661) 276-5714  
 Shaun C. McWherter, DFRC, Code RC, (661) 276-2530

# **AUTOMATIC COLLISION AVOIDANCE TECHNOLOGIES FLIGHT TESTS**

## **Summary**

The Automatic Ground Collision Avoidance System (Auto-GCAS) is designed to predict an impending collision with terrain and to engage an automatic pilot system to automatically perform a collision avoidance maneuver. The main goal is for the system to provide mishap protection for pilots that have lost situational awareness, become disoriented or mis-oriented, or are experiencing *g*-induced loss of consciousness (GLOC). The Auto-GCAS compares predicted fly-up trajectories with an estimated terrain profile to detect impending terrain collisions. If an impending collision is detected, the flight control system performs an automatic evasion maneuver by rolling to wings-level and pulling up to avoid a ground collision.

The Automatic Collision Avoidance Technology (ACAT) program is being conducted under a joint Memorandum of Agreement (MOA) among the Office of the Under Secretary of Defense for Personnel and Readiness (OUSD-P&R), the Air Force Research Laboratory Air Vehicles Directorate (AFRL/RB), and the National Aeronautics and Space Administration (NASA) Dryden Flight Research Center (DFRC) (Edwards, California). The DFRC is the designated Responsible Test Organization (RTO) for flight-testing the Auto-GCAS.

## **Objective**

The ACAT program was initiated to provide a technology solution for aviation to a Secretary of Defense mandated 75-percent reduction of Department of Defense (DoD) mishaps. Although this mandate was established at a higher level than the ACAT project, the results of the project are expected to contribute significantly to meeting that higher-level goal. The Fighter Risk Reduction Project is developing and demonstrating a modular software architecture that will facilitate integration onto fighter platforms, specifically the F-35, F-22, and F-16 fighter aircraft (all of the Lockheed Martin Corporation, Bethesda, Maryland).

Auto-GCAS is not a new technology, but has been matured to a comparatively high level on previous F-16 flight-test programs. The previous tests conducted over 2,000 automatic recoveries, validating the overall design approach and determining nuisance criteria. Previous tests also demonstrated that Auto-GCAS could have prevented nearly all of the historical F-16 Class A mishaps involving terrain impact, and confirmed operational applicability. The current Auto-GCAS design has been updated to incorporate a modular architecture critical for general applicability to other aircraft and includes a number of algorithm improvements.

The current flight tests have been organized to obtain results relative to the top four guiding requirements of the Auto-GCAS design:

1. Do no harm.
2. Do not impede mission performance.
3. Avoid ground collisions.
4. Collect data to facilitate transition of Auto-GCAS technology onto other platforms.

## **Approach**

The flight test objectives were developed to correlate with the guiding requirements. The top-level flight-test objectives are:

1. Evaluate Auto-GCAS with respect to failsafe operations.
2. Evaluate the mission utility of Auto-GCAS.



3. Evaluate the ability of Auto-GCAS to prevent ground collisions.
4. Evaluate Auto-GCAS for F-16 leave-behind capability.
5. Collect data to validate program requirements.

These objectives are being met with carefully selected flight-test points, which verify that the system does not interfere with standard aircraft operations, maintain the aircraft within its flight envelope, and prevent collisions with the ground. They also include in-flight simulation of historical F-16 mishap scenarios, which demonstrate that the system could have prevented the mishaps.

### **Status**

The ACAT project has completed 20 flight tests including CAT I (clean and utility) and CAT III (heavy and asymmetric) aircraft configurations. Many of the resulting findings have contributed to refining the system design. These findings are being implemented in a new version of the test-flight software, which will begin flight-testing in October of 2009. Figure 1 is an example of part of the flight-test data analysis being conducted. The green represents Digital Terrain Elevation Data (DETD®) (National Geospatial-Intelligence Agency, Bethesda, Maryland), which the system uses to determine ground proximity. The blue represents the predicted escape trajectory, including uncertainty. Figure 1 also indicates that the predicted trajectory successfully clears the terrain, implying that the aircraft would be saved from a collision.

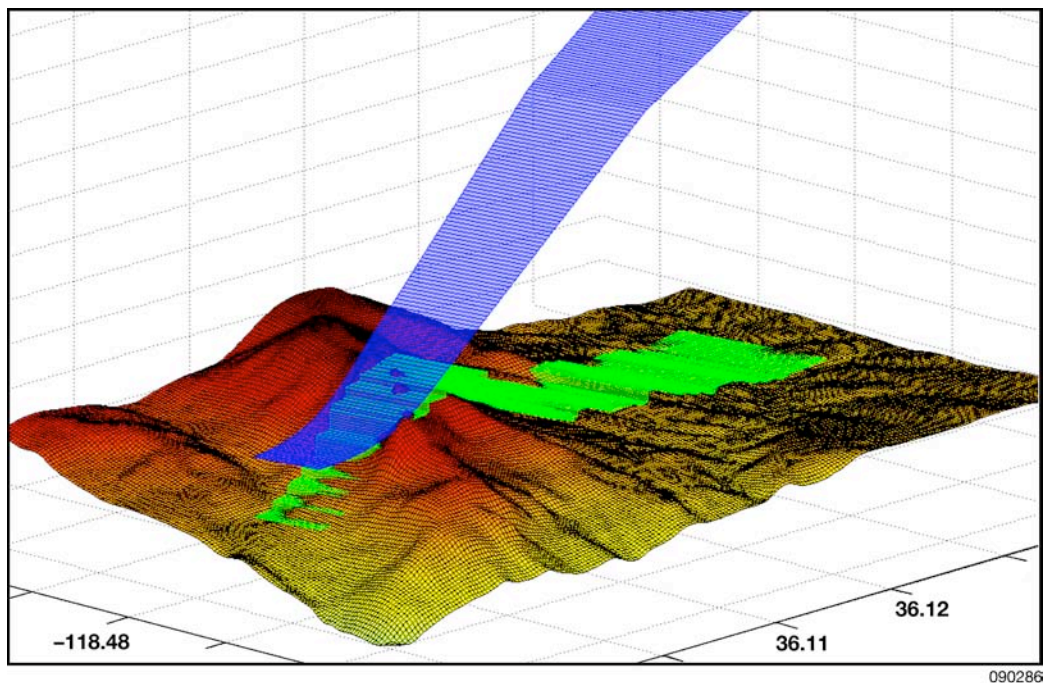


Figure 1. Auto-GCAS predicted trajectory and thinned Digital Terrain Elevation Data plotted atop simulated terrain.

### **Contacts**

Jack Ryan, DFRC, Code RC (661) 276-2558  
Mark Skoog, DFRC, Code Z (661) 276-5774  
Paul Sorokowski, AFFTC, (661) 276-3658



# EXTENSION OF KO STRAIGHT-BEAM DISPLACEMENT THEORY TO THE DEFORMED SHAPE PREDICTIONS OF CURVED STRUCTURES

## Summary

The Ko displacement theory originally developed for straight beams was applied to the deformed shape predictions of curved beams with different curvatures. The bending strains to be measured at equally spaced strain-sensing stations along the fiber optic strain sensor lines were generated from the finite-element analysis. The strains data was then input to the Ko straight-beam deflection equations for the calculations of deflections for the curved beams. The curved-beam deflections calculated from the Ko displacement theory were found to be slightly larger than the deflections calculated from the finite-element analysis. The deflection prediction error was found to increase progressively with the increasing beam curvature. Mathematical functions for curvature-effect corrections were established empirically, and were incorporated into the existing Ko straight-beam deflection equation. The resulting modified Ko displacement equation was found to be able to predict fairly well the deformed shape of the curved beams up to 90-deg arc. For the two-point supported curved beam of 22-deg arc cut out along the diameter of a generic Crew Exploration Vehicle (CEV) spherical shell, the Ko straight-beam theory was found to provide sufficiently accurate shape predictions without using the curvature-effect corrections.

## Objective

The objective is to examine the accuracy of the Ko straight-beam displacement equations as applied to the shape predictions of curved beams, and to explore the mathematical functions for curvature effect corrections.

## Approach

Curved beams with azimuth angles (0, 22.5, 45, 67.5, 90) deg were analyzed. Surface strains of the curved beams were generated from the finite-element analysis. The strain data was then input to the Ko straight-beam displacement theory for the shape predictions of curved beams. Based on the prediction differences between the finite-element analysis and the Ko displacement theory, mathematical correction functions were established to modify the Ko straight-beam displacement theory for the curved beam shape predictions (figs. 1 and 2).

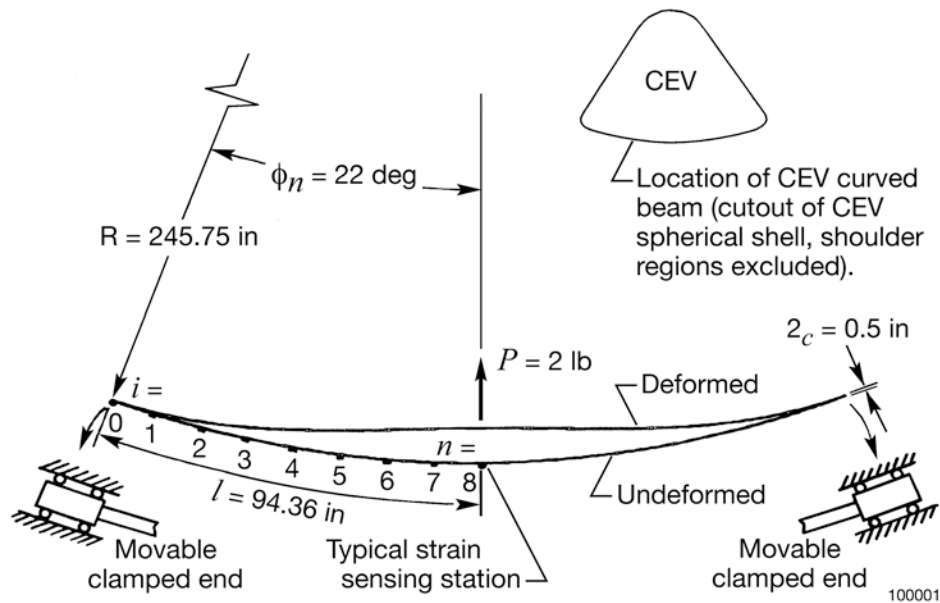


Figure 1. Two-point supported CEV curved beam under clamped support condition.

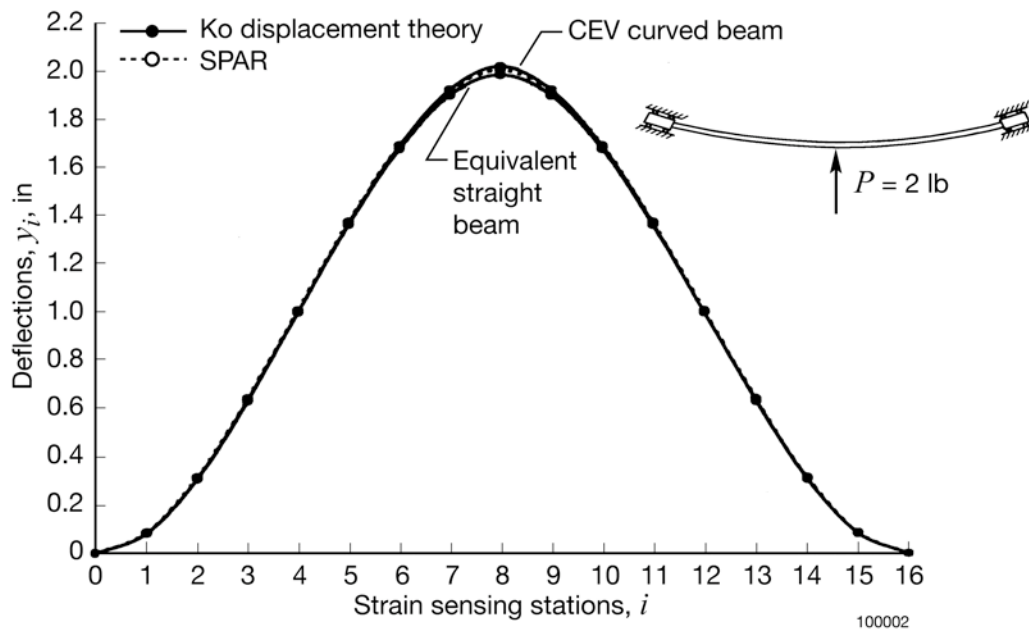


Figure 2. Comparisons of deflections calculated from SPAR program and Ko displacement theory for two-end clamped CEV curved beam and equivalent beam.

### Status

The modified Ko deflection equations with introduced curvature effect were found to provide reasonably accurate shape predictions for any curved beam up to a 90-deg arc. More refined curvature effect correction functions are being explored toward improving the accuracy of prediction.

### Contacts

Dr. William L. Ko, NASA DFRC, Code RS, (661) 276-3581  
 Dr. W. Lance Richards, NASA DFRC, Code T, (661) 276-3562

# APPLICATION OF KO DISPLACEMENT THEORY TO DEFORMED SHAPE PREDICTIONS OF DOUBLE-TAPERED IKHANA WING

## Summary

The Ko displacement theory formulated for non-uniform cantilever beams was applied to the shape prediction of the Ikhana wing. The wing deflection equations are expressed in terms of surface bending strains. Two multiplexed (Bragg gratings) fiber optic strain sensor lines are installed on the upper surface of the Ikhana wing to measure the bending strains at a number of equally spaced strain-sensing stations. The bending strain data can then be input to the two sets of deflection equations for the calculations of slopes, deflections, and cross-sectional twist angles of the Ikhana wing for generation of the overall deformed shapes of the entire Ikhana wing. The displacement equations developed were successfully validated for their accuracy by the finite element analysis. The Ko displacement theory was found to provide accurate deformed shape (bending and torsion) predictions of the Ikhana wing. The displacement equations combined with the fiber optic strain sensing system form a powerful tool for in-flight deformed shape monitoring of unmanned flexible aircraft wings. The calculated displacement data could ultimately be visually displayed before the eyes of the ground-based pilot to monitor the in-flight deformed shape of the wings of the unmanned aircraft.

## Objective

By installing highly multiplexed (Bragg gratings) fiber optic multiple strain sensors at discrete sensing stations on the Ikhana wing, it is possible to use the measured surface bending strain data to input to the Ko displacement equations to calculate the deflections and cross-sectional rotations of the Ikhana wing during flight. The purpose is first to predict the pre-flight deformed shapes of the Ikhana wing.

## Approach

The formulation of the Ko displacement theory for the non-uniform beams is based upon the modified beam differential equation. Using a piece-wise linear assumption and divide the beam domain into  $n$  sections, the beam slope and deflection equations for each beam section were then formulated in term of measured strains at  $n + 1$  strain sensing stations at the upper surface of the Ikhana wing (figs. 1 and 2).

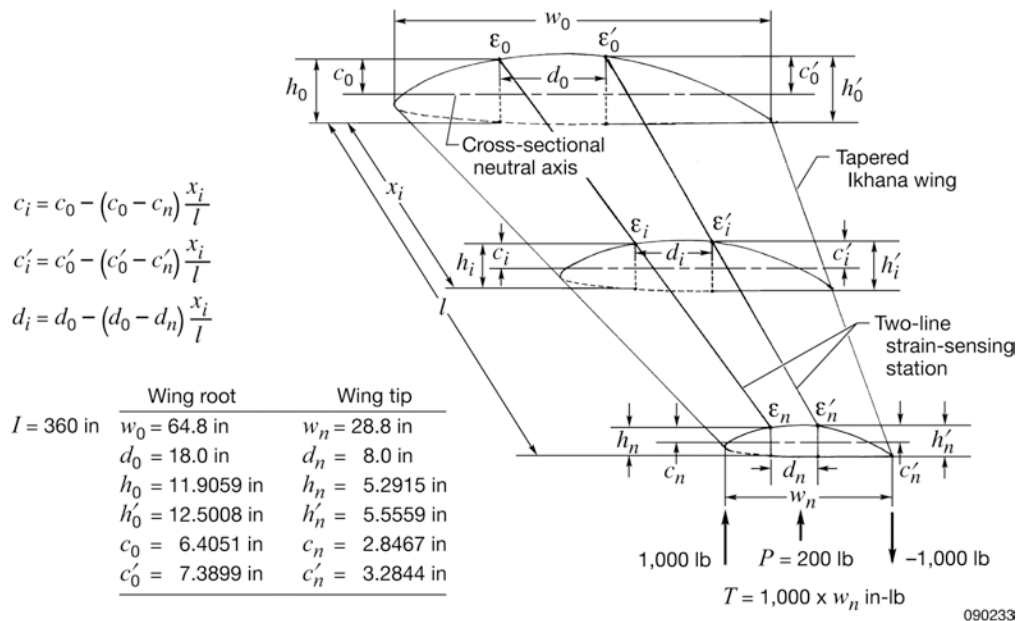


Figure 1. Double-tapered Ikhana wing installed with two strain sensing lines on the wing upper surface.

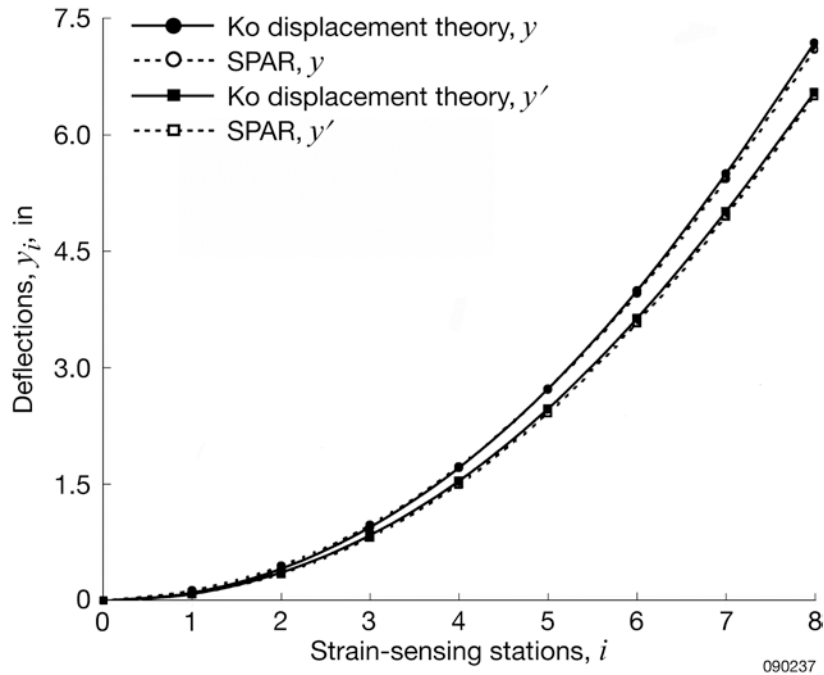


Figure 2. Comparison of Ikhana wing deflections calculated from the Ko displacement theory and Structural Performance and Resizing program.

### Status

The Ko displacement theory for non-uniform beams is being analytically validated with the aid of the Structural Performance and Resizing (SPAR) finite-element computer program. Cases tested were 1) tapered tubular cantilever beams, 2) un-swept and swept tapered wing boxes, 3) trapezoidal plates, 4) uninhabited aerial vehicle (UAV) wing, cantilever and two-point supported curved beams. The results show a high degree of accuracy of the Ko displacement theory in the structural deformed shape predictions. The Ko displacement theory and the associated fiber optics strain sensing system form a powerful tool for monitoring the in-flight deformed shapes of the aircraft wings. This innovative method, “Method for Real-time Structural Shape-Sensing” is patented (U.S. Patent No. 7,526,176, issued April 21, 2009).

### Contacts

Dr. William L. Ko, NASA DFRC, Code RS, (661) 276-3581  
 Dr. W. Lance Richards, NASA DFRC, Code T, (661) 276-3562

# **HANDLING QUALITIES PREDICTION OF AN F-16XL-BASED REDUCED SONIC BOOM AIRPLANE**

## **Summary**

A major goal of the Supersonics Project under the National Aeronautics and Space Administration (NASA) Fundamental Aeronautics program is sonic boom reduction of supersonic aircraft. An important part of this effort is the development and validation of sonic boom prediction tools to be used in aircraft design. As part of this effort a NASA Research Announcement (NRA) was awarded to The Boeing Company (Chicago, Illinois) to design modifications to reduce the sonic boom of the Lockheed Martin (Bethesda, Maryland) F-16XL airplane, tail number 849, operated by the NASA Dryden Flight Research Center (DFRC) (Edwards, California). The NRA covered the design of modifications to reduce the sonic boom as well as the development of design tools to assist in this effort. Two options for follow-on work included building and testing a wind-tunnel model of the modified F-16XL airplane, and airplane modifications and flight tests.

Part of this task was predicting the handling qualities of the modified airplane. Due to the high cost of modifying the existing F-16XL control laws, it was desirable to find a modification that reduced the aircraft sonic boom but did not degrade baseline aircraft handling qualities, allowing for the potential of flight-testing without changing the current control laws.

## **Objective**

The primary objective of this work was to determine an aerodynamic and mass properties envelope of the F-16XL airplane that would show aircraft designers to what extent the airplane could be modified without affecting the baseline handling qualities. The designers could then use this envelope to determine the effect of proposed modifications on aircraft handling qualities.

## **Approach**

A two-part approach was taken: 1) validation of the existing NASA DFRC F-16XL simulator and handling qualities tools; and 2) modification of the simulation to represent the modified airplane and determination of the modification envelope that would show how much the aircraft could be modified without affecting the baseline aircraft handling qualities.

The simulation was validated by feeding the pilot inputs from flight-test data into the simulation. The resulting aircraft responses were compared with the flight-test response of same pilot input. The results showed that simulator response matched the flight-data. The handling qualities tools were also validated using a similar method; comparing the prediction to the results from the original flight-test program. As with the simulation, the handling qualities results matched those in the report from the original flight-test program.

With the simulation and handling qualities tools validated, the simulation was modified to represent potential aerodynamic and mass properties changes of the modified aircraft. The modification parameters were tested individually. The pitch frequency sweeps were used as the simulation input. Various handling quality analysis tools were then used on the recorded results for handling quality prediction. In addition to computational metrics, Cooper-Harper ratings were collected from several pilots performing a 1-g tracking task. A tracking task is shown in figure 1 and performance criteria are defined in table 1.

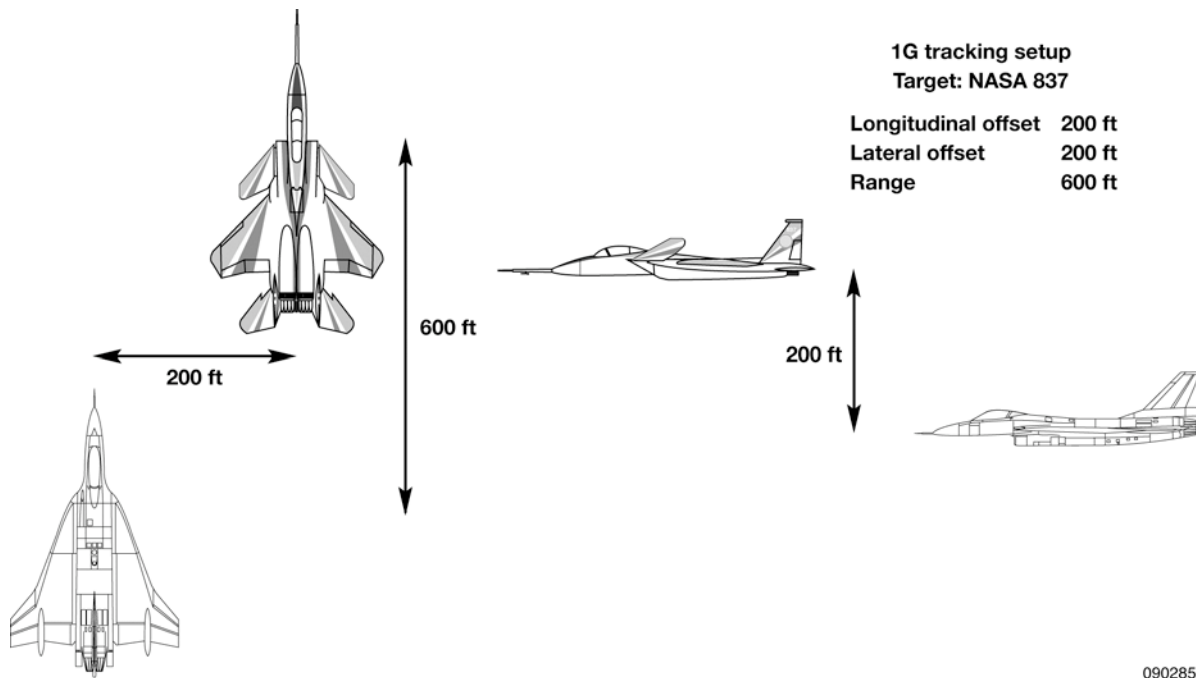


Figure 1. Pilot Cooper-Harper ratings evaluation.

Table 1. Performance criteria.

Desired	Adequate
Acquire gross lock within 3 sec	Acquire gross lock within 5 sec
1 overshoot	2 overshoots
No pilot-induced oscillation	No pilot-induced oscillation
Hold lock for 10 sec	Hold lock for 10 sec

Each pilot flew the baseline aircraft and several modified aircraft at two or three of the flight conditions. Cooper-Harper Ratings (CHR), pilot-induced oscillation (PIO) ratings, and pilot comments were obtained for each configuration and condition. Generally, pilot ratings agreed with the predicted ratings, although there were some discrepancies due to lack of physical cues. Only the longitudinal axis was investigated.

### Status

The current NRA task was completed in mid 2009. The task was not extended, as the predicted sonic boom magnitude could not be reduced to meet project goals. There were also concerns about supporting operation of the F-16XL aircraft.

A NASA Technical Memorandum will be written as well as an American Institute of Aeronautics and Astronautics (AIAA) technical paper. Some further simulation testing is also planned in the near future.

### Contacts

Bruce R. Cogan, DFRC, Code RC, (661) 276-2627  
 Seung Y. Yoo, DFRC, Code RC, (661) 276-5247

# **SOFIA CLOSED– AND OPEN–DOOR AERODYNAMIC ANALYSES**

## **Summary**

Extensive work has been undertaken to prepare for open-door flight tests of the Stratospheric Observatory for Infrared Astronomy (SOFIA) Platform project. Work completed includes aerodynamic model development through parameter estimation from flight data collected during closed-door flight tests and baseline flight tests, tuft analysis from the closed-door flight tests, and simulation development. Analysis tools were developed to allow efficient processing of data from the upcoming open-door flights. Additional analysis tool development is ongoing.

## **Objective**

To determine the aerodynamic characteristics of the SOFIA airplane in the closed-door configuration and prepare for gathering aerodynamic data during the open-door configuration flight phase.

## **Approach**

A set of diverse tasks was undertaken to prepare for the upcoming open-door flights of the SOFIA 747SP (The Boeing Company, Chicago, Illinois) airplane (fig. 1). Significant effort was devoted to completing the evaluation of the aerodynamic characteristics of the modified 747SP airplane in its closed-door configuration. Aerodynamic models of the airplane were developed using a combination of results from parameter estimation techniques, from existing closed-door and baseline flight data, and from existing aerodynamic models. During the closed-door flights, the aft portion of the fuselage was tufted and video was taken from a chase plane. This video was analyzed for various flight conditions, and general flow descriptions for the aft fuselage of the 747SP were developed.



ED07-0079-02

Figure 1. SOFIA 747SP airplane.

Several tools were developed to prepare for open-door flight tests. An engineering simulation was developed using Simulink® (The MathWorks™, Inc., Natick, Massachusetts) to validate the existing aerodynamic models and aid in development of improved open-door configuration aerodynamic models. The simulator will also be used to simulate proposed flight-test maneuvers and evaluate their effectiveness. The aerodynamic models developed for this simulator will be incorporated into the SOFIA Platform project simulation. Methods to automate tufting analysis using the planned on-airplane video systems were investigated. These methods would involve using image recognition software to identify the tuft behavior over a set period of time. The data would be compiled and a flow map could be created for a given flight condition.

### **Status**

The analysis of data from the SOFIA closed-door flights is complete. Work is continuing in order to support upcoming open-door flight tests.

### **Contacts**

Stephen Cumming, DFRC, Code RA, (661) 275-3732  
Mike Frederick, DFRC, Code RA, (661) 276-2274  
Mark Smith, DFRC, Code RA, (661) 276-3177



# **X-48B AERODYNAMIC SYSTEM IDENTIFICATION**

## **Summary**

The X-48B project is a joint partnership between the National Aeronautics and Space Administration (NASA), the Air Force Research Laboratory (AFRL), and The Boeing Company (Chicago, Illinois). This next-generation research aircraft is an 8.5-percent dynamically scaled blended wing body aircraft with a unique structural configuration and outer mold line, and is controlled using 20 aerodynamic control surfaces.

## **Objective**

The objectives particular to the Research Aerodynamics and Propulsion Branch (Code RA) of the NASA Dryden Flight Research Center (Edwards, California) include the estimation from appropriately designed flight-test maneuvers of the open-loop stability and control derivatives for the X-48B aircraft over a range of angles of attack and sideslip below the stall boundary. Particular emphasis is placed on estimating the change in the control surface effectiveness when surfaces are moved singularly or moved in concert with adjacent surfaces. Finally, the estimated values of the derivatives will be compared with preflight predictions, based on wind-tunnel data and computational fluid dynamics, and as implemented in the Boeing simulation.

The primary research goals are as follows:

1. Assess the applicability and utility of existing system identification test techniques and analysis methods to an aircraft with a large number ( $>10$ ) of control effectors, significant stability augmentation, low wing loading, and an airdata system that is embedded in significant local flow.
2. Develop relevant experience with the isolated issues of low wing loading and an airdata system that is embedded in significant local flow using previously acquired flight data from a comparable configuration; U.S. Air Force TG-14 glider (Aeromot, Rio Grande do Sul, Brazil).
3. Perform system identification on the X-48B aircraft and compare the stability and control derivatives estimated from flight data with those of the wind-tunnel testing and the simulation aerodynamic database.
4. Publish and present research findings.

These objectives support the eventual goal of increased performance in efficiency and maneuverability for hybrid-wing-body configurations. The research performed may also enhance the capability to extract stability and control derivative estimates from flight data for air vehicles with low wing loading, high-control-surface-count, and significant control augmentation.

## **Approach**

Obtaining consistent stability and control derivative estimates for the X-48B will prove challenging owing to the multiple feedback paths to the multiple control effectors. Multiple approaches will be used to address the challenge of multiple correlated control inputs.

The existing X-48B simulation will be utilized as an evaluation tool, and as a gauge of the utility and validity of the analysis tools. By applying the analysis tools to parameter identification (PID) maneuvers flown in the simulation, for which the stability and control derivatives are (in principle) known, the performance of the tools can be evaluated. Only those tools and techniques which perform well with the simulated data will be applied to the actual flight-test data. Furthermore, the simulation will be used to select the maneuver amplitude for the flight-test excitation.

At least three techniques will be evaluated to address the challenge of correlated control motions: single-surface and paired-surface (symmetric and antisymmetric) excitation; multi-maneuver analysis for de-correlation; and multisine excitation for de-correlation.

For the first technique, all control surface derivatives will be fixed at their aerodynamic database (predicted) values, and control surface derivatives will be estimated only for those surfaces (or surface pairs), which are directly excited. The quality of the resulting time history fit and the maneuver-to-maneuver consistency of the estimates will be the primary metric by which the results will be judged.

The second technique will involve data mining to identify and collect maneuvers from different portions of a flight or even from multiple flights. In this approach, the objective is to collect into one super maneuver enough data so that the collective control surface motions are largely uncorrelated, or have significant uncorrelated portions. Through the technique of multisine inputs, many surfaces are excited simultaneously with distinct frequencies to facilitate matching of particular surfaces to particular portions of the dynamic response of the aircraft. While this is a more complicated analysis technique, it may solve the correlated-controls issue without imposing artificial constraints on control surface derivative values.

The third technique will involve grouping the 20 control surfaces into symmetric and antisymmetric groups, in order to reduce the number of independent excitations to a manageable number. Multisine excitations, which are inherently uncorrelated, will be developed for application to the control-surface groups in flight test. Multiple control-surface groups have been defined, including some groups for which multiple control surfaces are moved in concert with their neighboring surfaces, potentially allowing for identification of control-surface effectiveness amplification in such a scenario. This third technique holds the most promise for productive and efficient use of X-48B flight-research resources.

### **Status**

Researchers have completed the development, modification, or validation of most of the computational and analysis tools required for PID on the X-48B blended wing body aircraft. The team is now analyzing open- and closed-loop PID maneuvers in the Boeing nonlinear simulation, using test cases of varying degrees of complexity. Based on the results of these analyses, a set of flight research test points will soon be defined that will allow the team to begin performing system identification on actual flight data, and working toward the ultimate objective of aerodynamic system identification of the aircraft.

### **Contacts**

Nalin A. Ratnayake, DFRC, Code RA, (661) 276-5876  
Brian R. Taylor, DFRC, Code RA, (661) 276-5451

# **ADAPTIVE FEEDFORWARD CONTROL FOR GUST LOADS ALLEVIATION, MODAL SUPPRESSION, AND FLUTTER/LIMIT CYCLE OSCILLATION PREVENTION**

## **Summary**

An Adaptive Feedforward/Feedback Control (AFFC) framework is being developed to suppress aircraft structural vibrations and to increase the resilience of the flight control law in the presence of aeroelastic/aeroservoelastic (AE/ASE) interactions. Specifically, the adaptive feedforward controller is designed to reduce any atmospheric induced structural vibrations of the aircraft. The adaptive feedback controller is applied to suppress any AE/ASE interactions, and prevent the onset of Flutter/Limit Cycle Oscillation (LCO) instabilities within the flight envelope of a flexible aircraft.

## **Objective**

The overall objective is to develop new techniques and methodologies to enhance aerospace flight controls systems by providing tools for in situ identification, control adaptation, and feedforward algorithms on the basis of the success of the AFFC framework to facilitate or suppress structural vibrations due to AE/ASE interactions. The new tools for the enhancement of the flight controls systems are being developed as an embedded software package in the form of a MATLAB® (The MathWorks™, Inc., Natick, Massachusetts) aerospace block-set library, called In-situ Adaptive Feedforward/Feedback Aeroservoelastic Control (i-AFFAC) block-set algorithms for robust on-line adaptation of flight controls systems.

## **Approach**

The i-AFFAC block set consists of four main modules that cover data based modeling (identification); adaptive feedforward algorithms; adaptive feedback algorithms; and demonstration cases that illustrate how to use the i-AFFAC.

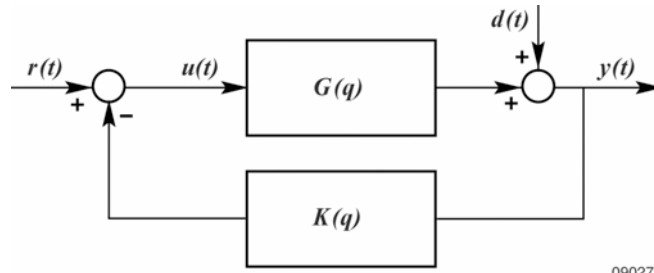
The in-situ identification module consists of three system identification techniques: PolyMAX (LMS International, Leuven, Belgium) methodology, a step-based realization algorithm, and a 3-2-1-1 excitation-based realization algorithm.

The adaptive feedforward algorithm module consists of recursive least-square algorithms, a basis function generator, Orthonormal Finite Impulse Response (ORTFIR) filtering, and gust sensor models.

The adaptive feedback algorithm consists of additive feedback perturbation control, coprime feedback perturbation control, optimal robust dynamic output feedback control, and optimal robust static output feedback control (fig. 1).

The demonstration case module consists of a damaged/undamaged F/A-18 (McDonnell Douglas, now The Boeing Company, Chicago, Illinois) Active Aeroelastic Wing (AAW) nonlinear aeroelastic system and a Northrop Grumman Corporation (Los Angeles, California) Sensor Craft wind-tunnel model for demonstration purposes.

The method for adaptive feedback control design for flutter boundary expansion is to directly estimate the open-loop unstable plant from the closed-loop experiment using coprime factorization and design the optimal control based on the estimated open-loop unstable model. The iteration starts from the controller design for an open-loop stable plant, and the iteration process can proceed until a better controller is not available.



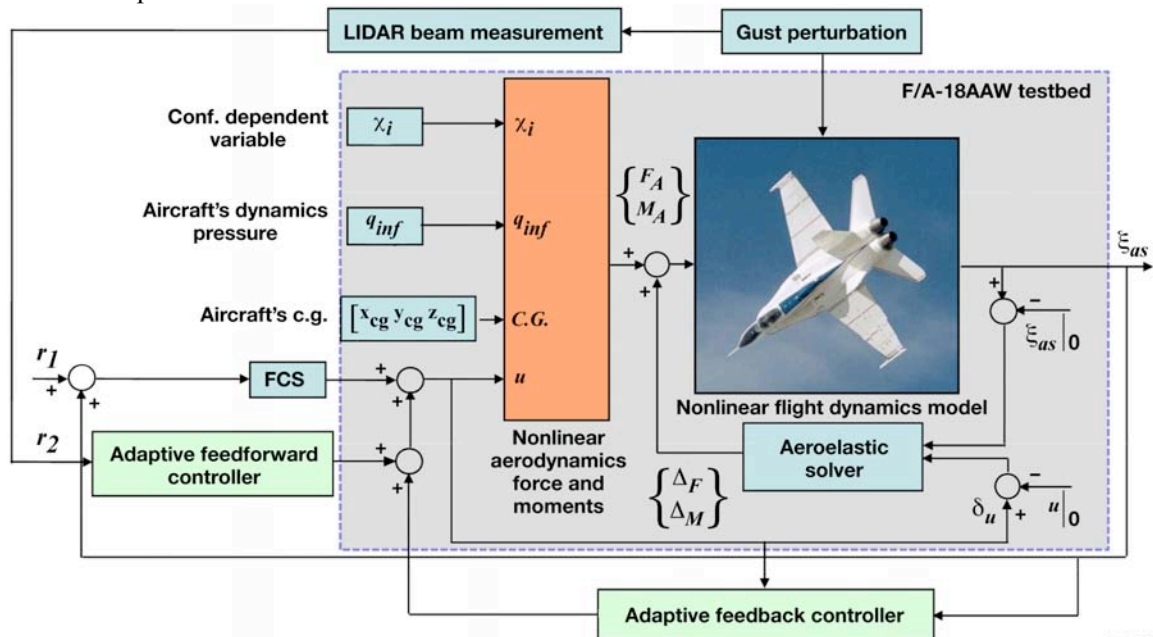
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Figure 1. Feedback control with disturbance.

### Status

Current work consists of the development of the nonlinear 6-degree-of-freedom flexible flight dynamics of the F/A-18 AAW including the gust excitation effects; the preliminary development of an adaptive feedforward control algorithm for gust-induced structural vibration of the aircraft; the preliminary development of an adaptive additive perturbation feedback control algorithm to suppress the AS/ASE vibrations; the development of adaptive coprime feedback control algorithms to extend the flutter boundary; and the evaluation of the proposed adaptive feedforward/feedback control algorithms by applying them to the linear/nonlinear F/A-18 AAW aeroelastic Simulink® (The MathWorks™, Inc., Natick, Massachusetts) model (fig. 2).

Future work will include developing algorithms for on-line frequency response function/auto/cross spectrum estimation and time response subspace identification; modifying the frequency domain PolyMAX method for the on-line modal estimation; writing MATLAB® M/C-MEX S-functions for on-line physical mode estimation; and developing on-line algorithms suitable for implementation on the NASA F/18 airplane tail number 853.



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Figure 2. Adaptive feedback/feedforward control scheme.

### Contacts

Martin J. Brenner, DFRC, Code RS, (661) 276-3793  
 Jie Zeng, ZONA Technology, Inc., Scottsdale, Arizona, (480) 945-9988  
 Professor Raymond de Callafon, University of California at San Diego, (858) 534-3166

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“Smart Material Coated Fiber Bragg Grating Sensors,” Provisional Patent Application filed July 21, 2009 by Michael Emmonds (SSC), Lance Richards, Mohanchandra Panduranga (SSC), Greg Carman (SSC), and Sunny Karnani (SSC).



## **NASA Medals**

### ***Exceptional Achievement Medals***

Jennifer H. Cole  
Edward A. Haering  
James A. Lee  
Ting C. Tseng

### ***Exceptional Engineering Achievement Medal***

Allen Parker, Jr.

### ***Exceptional Administrative Achievement Medal***

Everlyn Y. Cruciani

### ***Exceptional Public Service Medal***

Florence B. Norman

### ***Group Achievement Awards***

Fiber Optic Shape Sensing Team  
Intelligent Flight Control System Project Team  
Gulfstream III UAVSAR Project Team

### ***Federal Laboratory Consortium – Interagency Partnership Award***

Wildfire Research and Demonstrated Control Applications; USDA Forest Service; FAA, Ames Research Center, Dryden Flight Research Center.

## **Fiscal Year 2009 Infusion Successes**

Tao Systems, Inc., “Distributed Aerodynamic Sensing and Processing (DASP) Toolbox (FY07 Seed Fund Project).

Rolling Hills Research Corp., “A Rayleigh Flow Experiment on N20 to assess the limits of N20 Cooling,” STTR Phase III.

## **Fiscal Year 2009 Partnership Summary**

“Highly Reliable Reusable Launch System (HRRLS),” Holleman AFM; Embry Riddle; General Atomics Inc., Dryden Flight Research Center, Goddard Space Flight Center (FY06 Seed Fund).

“Distributed Aerodynamic Sensing and Processing (DASP) Toolbox,” AFRL and NASA Memorandum of Agreement.



<b>REPORT DOCUMENTATION PAGE</b>					<i>Form Approved OMB No. 0704-0188</i>	
<p>The public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Department of Defense, Washington Headquarters Services, Directorate for Information Operations and Reports (0704-0188), 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302. Respondents should be aware that notwithstanding any other provision of law, no person shall be subject to any penalty for failing to comply with a collection of information if it does not display a currently valid OMB control number.</p> <p><b>PLEASE DO NOT RETURN YOUR FORM TO THE ABOVE ADDRESS.</b></p>						
<b>1. REPORT DATE (DD-MM-YYYY)</b> 01-05-2010		<b>2. REPORT TYPE</b> Technical Memorandum			<b>3. DATES COVERED (From - To)</b>	
<b>4. TITLE AND SUBTITLE</b> 2008 and 2009 Research and Engineering Annual Report				<b>5a. CONTRACT NUMBER</b>		
				<b>5b. GRANT NUMBER</b>		
				<b>5c. PROGRAM ELEMENT NUMBER</b>		
<b>6. AUTHOR(S)</b> Flick, Bradley C.; and Voracek, David F.				<b>5d. PROJECT NUMBER</b>		
				<b>5e. TASK NUMBER</b>		
				<b>5f. WORK UNIT NUMBER</b>		
<b>7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)</b> NASA Dryden Flight Research Center P.O. Box 273 Edwards, California 93523-0273					<b>8. PERFORMING ORGANIZATION REPORT NUMBER</b>  H-3027	
<b>9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)</b> National Aeronautics and Space Administration Washington, DC 20546-0001					<b>10. SPONSORING/MONITOR'S ACRONYM(S)</b>  NASA	
					<b>11. SPONSORING/MONITORING REPORT NUMBER</b>  NASA/TM-2010-214655	
<b>12. DISTRIBUTION/AVAILABILITY STATEMENT</b> Unclassified -- Unlimited Subject Category 99 Availability: NASA CASI (301) 621-0390      Distribution: Standard						
<b>13. SUPPLEMENTARY NOTES</b> Flick and Voracek, NASA Dryden Flight Research Center. An electronic version can be found at <a href="http://dtrs.nasa.gov">http://dtrs.nasa.gov</a> or <a href="http://ntrs.nasa.gov/search.jsp">http://ntrs.nasa.gov/search.jsp</a>						
<b>14. ABSTRACT</b> Selected research and technology activities at NASA Dryden Flight Research Center are summarized. These activities exemplify the Center's varied and productive research efforts.						
<b>15. SUBJECT TERMS</b> Aerodynamics, Aerostructures, Flight, Flight controls, Flight systems, Flight test, Instrumentation, Propulsion, Structural dynamics, Structures						
<b>16. SECURITY CLASSIFICATION OF:</b>			<b>17. LIMITATION OF ABSTRACT</b>	<b>18. NUMBER OF PAGES</b>	<b>19b. NAME OF RESPONSIBLE PERSON</b>	
<b>a. REPORT</b>	<b>b. ABSTRACT</b>	<b>c. THIS PAGE</b>			STI Help Desk (email: <a href="mailto:help@sti.nasa.gov">help@sti.nasa.gov</a> )	
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